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Space Shuttle Program

MSC-03332

IBM

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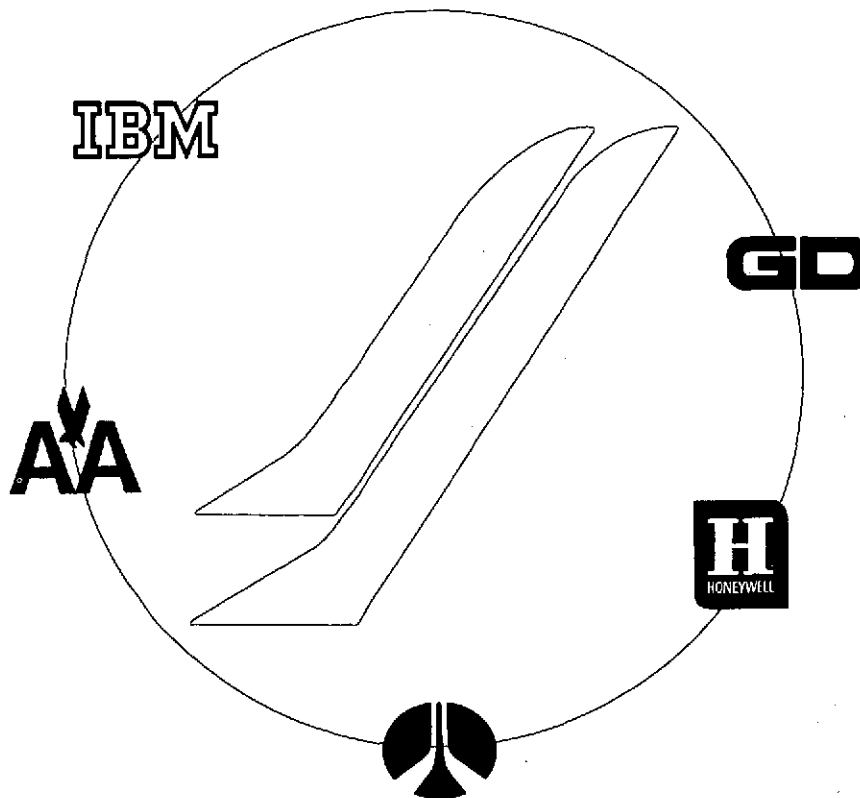
Space Shuttle Phase B"
Final Report

Volume II. Technical Summary

Contract NAS9-10960
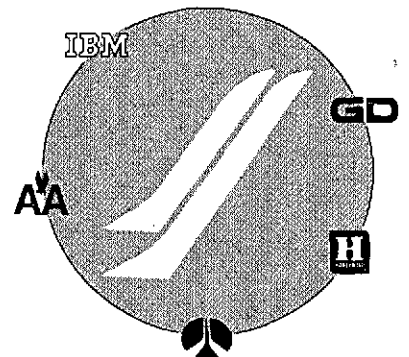DRL T-751, Line Item 6
DRD SE-420T
SD 72-SH-0012-2
15 March 1972

SPACE SHUTTLE

PHASE B'' FINAL REPORT

VOLUME II TECHNICAL SUMMARY

Approved by

B. Hello

B. Hello
Vice President and Program Manager
Space Shuttle Program

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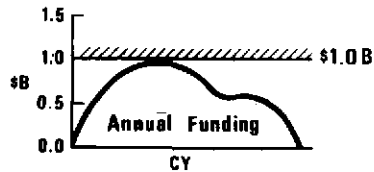
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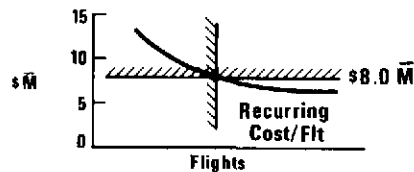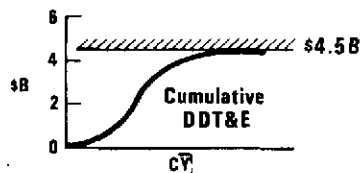
22SV17857A

The Space Shuttle Program must meet three funding requirements:
(1) $1 billion maximum annual funding, (2) a $4.5 billion cumulative design,
development, test, and evaluation (DDT&E) cost exclusive of NASA require-
ments, and (3) a recurring cost per flight no greater than $8 million. It will
be seen in the evaluation that no program satisfactorily met all of these gates.
Those which typically had a low cumulative DDT&E and/or annual funding
requirement generally exceeded the recurring cost per flight. And those
programs which exceeded the cumulative DDT&E and/or annual funding
requirement generally met the recurring cost per flight requirement.
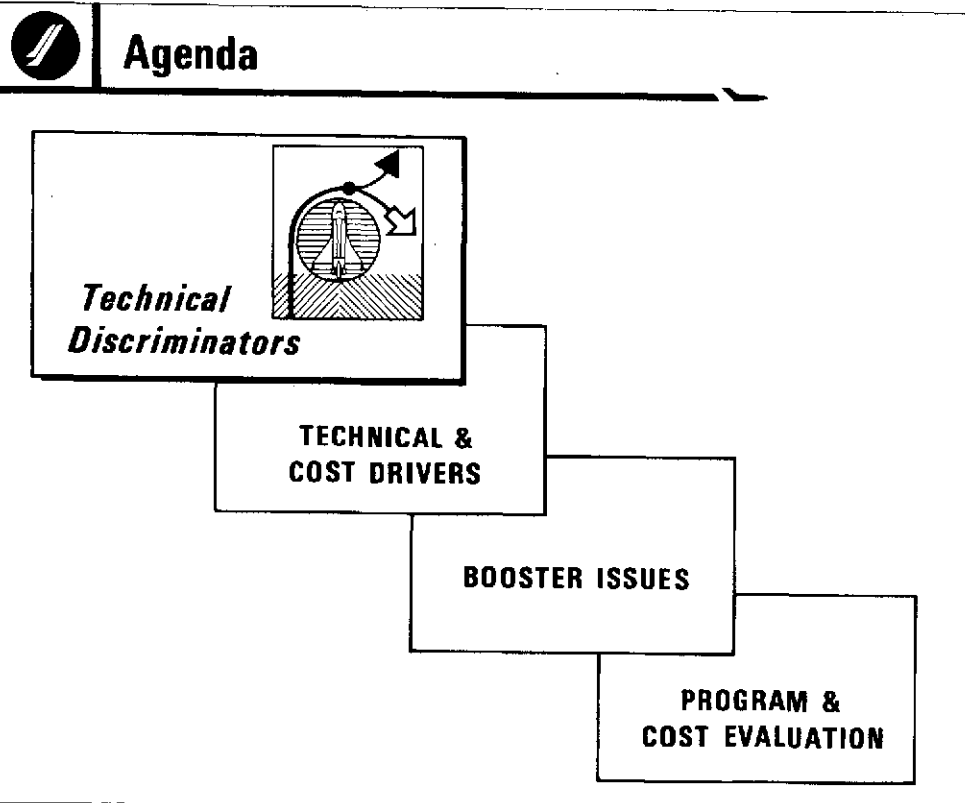
- 1 -

Agenda

- Technical Discriminators
- TECHNICAL & COST DRIVERS
- BOOSTER ISSUES
- PROGRAM & COST EVALUATION

22SV18069C

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Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
• SRM TVC Requirements	
• Stage Separation	
• SRM Thrust Termination Requirements	
ABORT	
• Mode	
• Control	
• MPS Requirements	
ENVIRONMENT IMPACT	
TEST & FACILITY IMPACT	

22SV18242B

The key technical issues to be discussed in the following pages are listed
in the chart. Dynamic studies were completed to determine the thrust
vector control requirements, separation modes, thrust termination
requirements, and abort modes for both series and parallel burn systems
and for both liquid and solid rocket motor boosters. The requirements for
off-the-pad and in-flight abort were determined, together with the main
propulsion system requirements to assure elimination of the down-range
landing requirement. An evaluation of the relative test and facility
requirements for liquid-fed versus solid propellant motor boost systems
was made, and the impact of the solid rocket motor on the vehicle and
ground environment was determined.

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Study Matrix

RECOVERABLE PFB

Series
15 × 60 Orbiter
14 × 45 Orbiter

Parallel
15 × 60 Orbiter

SRM'S

120 SRM's

Series
15 × 60 Orbiter

Parallel
15 × 60 Orbiter
14 × 45 Orbiter

156 SRM's

Series
15 × 60 Orbiter

Parallel
15 × 60 Orbiter
14 × 45 Orbiter

RECOVERABLE F-1 PUMP FED

Series
15 × 60 Orbiter
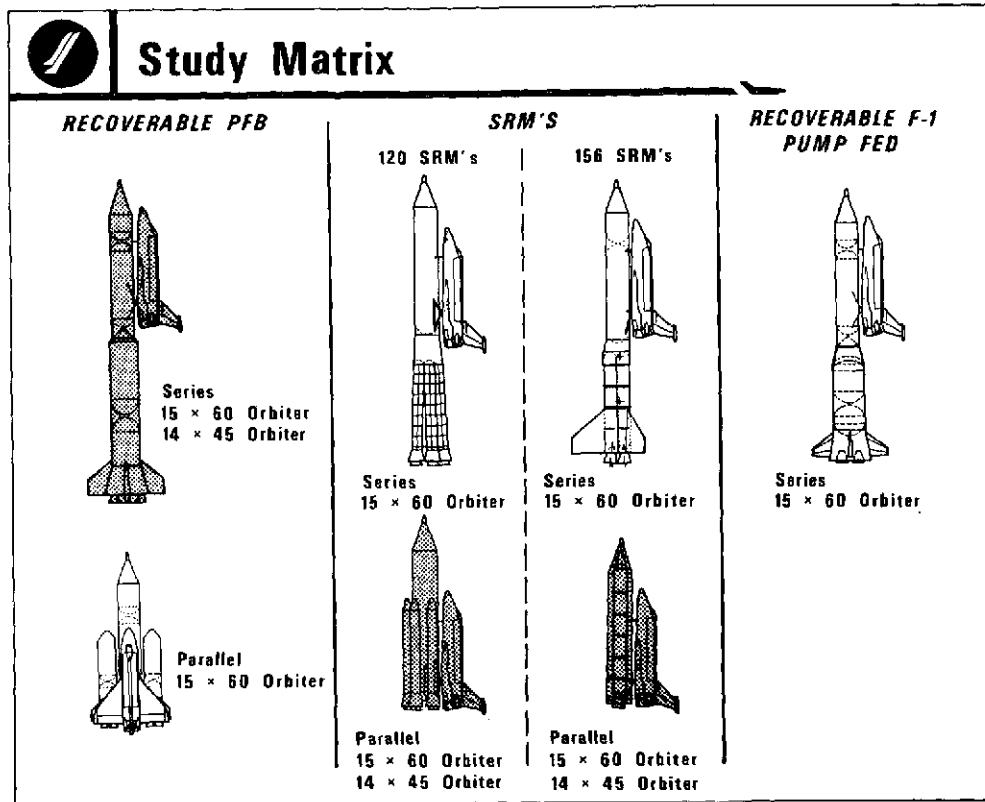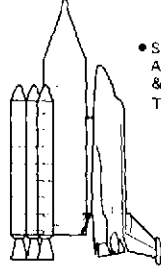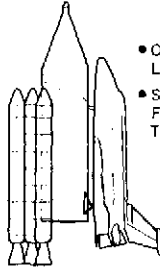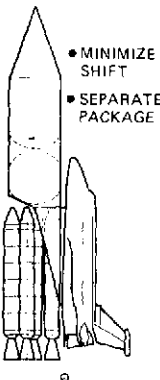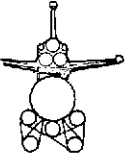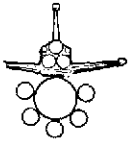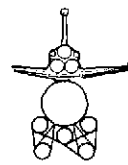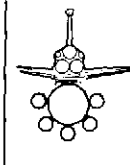
22SV18046C

On the opposite chart is shown the spectrum of launch configurations
analyzed in this study period. Study emphasis was placed on three
systems, the recoverable pressure-fed booster system in a series-burn
mode and two parallel burn systems, the first using 120-inch diam-
eter SRM's and the second using 156-inch-diameter SRM's. In each
case, two orbiter systems were analyzed, a 15-foot-diameter by
60-foot long cargo bay orbiter and a 14-foot-diameter by 45-foot
long cargo bay orbiter.

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- SRM STRUCT ATTACH FWD & AFT OF LH$_2$ TANK
- SEPARATE ONE PACKAGE

- ORB ATTACH AT LH$_2$ BULKHEADS
- SRMS LOCATED FOR AFT THRUST TERMINATION
- SEPARATE ON PACKAGE

- MINIMIZE Z GG SHIFT
- SEPARATE ONE PACKAGE

REQ 6 X 1206

CONTROL	128 IN. Z c.g. SHIFT	40 IN. Z c.g. SHIFT	131 IN. Z c.g. SHIFT	42 IN. Z c.g. SHIFT	35 IN. Z c.g. SHIFT
LOAD PATH	COMPLEX +15,000 LB	GOOD	COMPLEX +15,000 LB	GOOD	COMPLEX +10,000 LB
SEPARATION	COMPLEX MECHANISM	PARALLEL LINKS	COMPLEX MECHANISM	PARALLEL LINKS	DIFFICULT - STRUCT INTERFERENCE
THRUST TERMINATION	OK	FWD PORTS HAZARD	OK	AFT PORTS SAFEST	DIFFICULT

22SV18132B

Several parallel-burn 120-inch solid rocket motor configurations were examined initially. Various orientations of the solid rocket motors were investigated to minimize control requirements and to simplify the separation problem. Those arrangements which simplified the separation problem resulted in extremely high structural weights and at the same time vastly complicated the thrust vector control requirements. Also the axial location of the rocket motors was varied to obtain a reasonable location for thrust termination ports. It was determined that ports located at the aft end of each solid motor was the most attractive design. Therefore, of the configurations shown, the one with the aft located ports together with a cluster of five solid rocket motors mounted circumferentially around the external oxygen-hydrogen tank was the most attractive. In a subsequent chart, the configuration used for dynamic analyses will be shown. It differs slightly from that just described because it was selected early in the study prior to the configuration analysis illustrated on the opposite chart.

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156 in. SRM Parallel Burn

Alternate Configurations
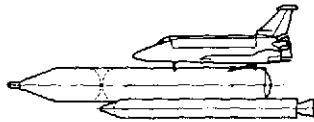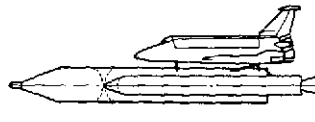
BOTTOM MOUNTED	BASELINE SIDE MOUNTED	AFT MOUNTED
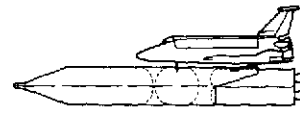
CONTROL	110 IN. Z c.g. SHIFT	34 IN. Z c.g. SHIFT	34 IN. Z c.g. SHIFT
LOAD PATH	COMPLEX +10,000 LB	GOOD	COMPLEX +4,000 LB
SEPARATION	GOOD LOCATION	PARALLEL LINKS	DIFFICULT STRUC- TURAL INTERFERENCE
THRUST TERM	GOOD LOCATION FOR AFT TT PORTS	GOOD LOCATION FOR AFT TT PORTS	GOOD ARRANGEMENT FOR AFT TT PORTS

22SV18143A

Alternate configurations of the 156-inch-diameter solid rocket motor
parallel-burn system were investigated with results similar to those for
the 120-inch system and for the same reasons. The most attractive sys-
tem again had the solid motors located somewhat aft to permit a safe
location for thrust termination ports, and with the motors located to mini-
mize the Z center of gravity shift during solid rocket motor operation.
This configuration arrangement is noted again on a later chart.

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PFB Series - Configuration Arrangement

GLOW (K Lb)	6.568
BLOW (K Lb)	5.254
OLOW (K Lb)	1.314
V$_s$ (FPS)	5.500

80 Ft

323 Ft

26 Ft Dia

300 in. Dia

164.4 Ft
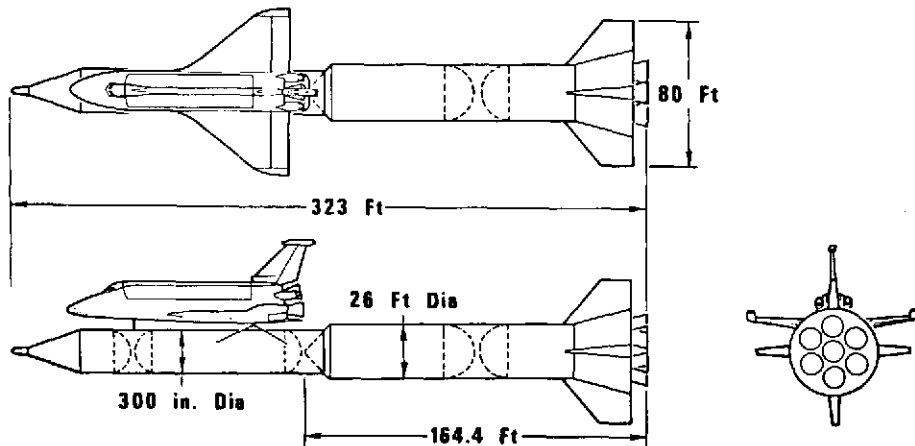
22SV17887B

The pressure-fed-booster series configuration arrangement is a conventional system with the thrust line of the boost motors close to the total system center of gravity. The separation system resembles the Saturn V/S-II dual plane system in concept. Fins on the booster are sized to provide adequate stability to minimize control requirements and flight aerodynamic loads.

- 9 -
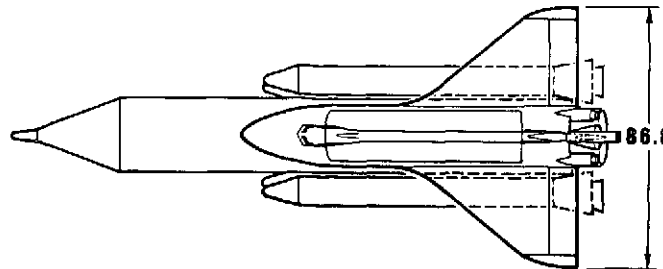
GLOW	(K Lb)	5.254
BLOW	(K Lb)	3.438
OLOW	(K Lb)	1.816
V_S	(FPS)	5.214

86.8

75 Ft

300 in. Dia

50°

112 Ft

184 Ft
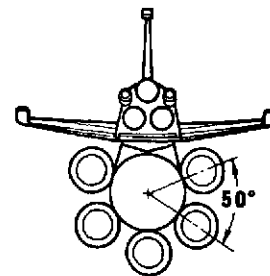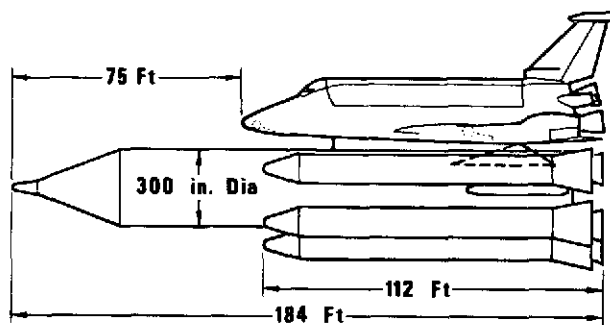
22SV17889B

The configuration used for control and separation studies is shown. The final recommendation has been described. In that configuration the solid motors are positioned farther aft relative to the tank to permit thrust termination port actuation at the aft end of the SRM's.

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Configuration Arrangement

156 Parallel

OLOW	1.785M Lb
BLOW	3.112
GLOW	4.898
V$_{Stage}$	5333 FPS

(2) - 156 in. SRM
2.939M Lb Thrust
Each

22SV18140B
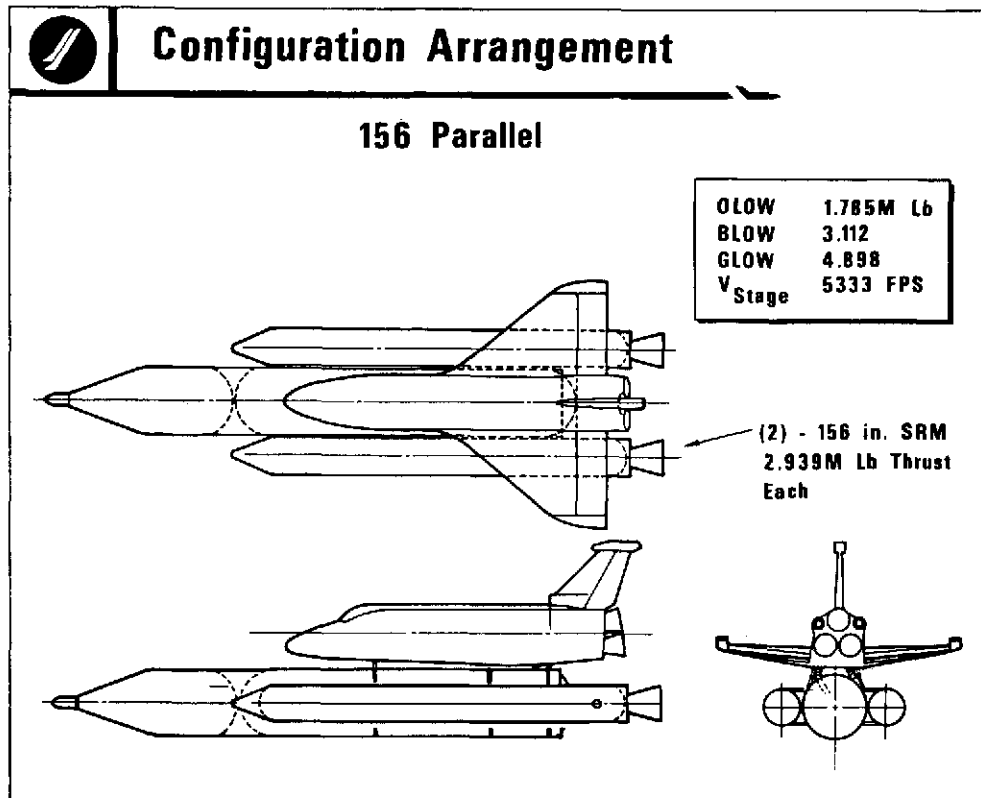
The illustration shows the system parameters for the selected configuration which has been described.

PITCH AERODYNAMIC CENTER & CENTER OF GRAVITY

AC ($C_N < 0$)

AC ($C_N > 0$)

MACH NUMBER

- ASSYMETRIC CONFIGURATION PRODUCES DUAL AC'S, F/C REQMT SET BY NEGATIVE α'S

YAW AERODYNAMIC CENTER & CENTER OF GRAVITY

c. g.

AC

MACH NUMBER

DATA BASE: MSFC TWT NO. 512

22SV18146A
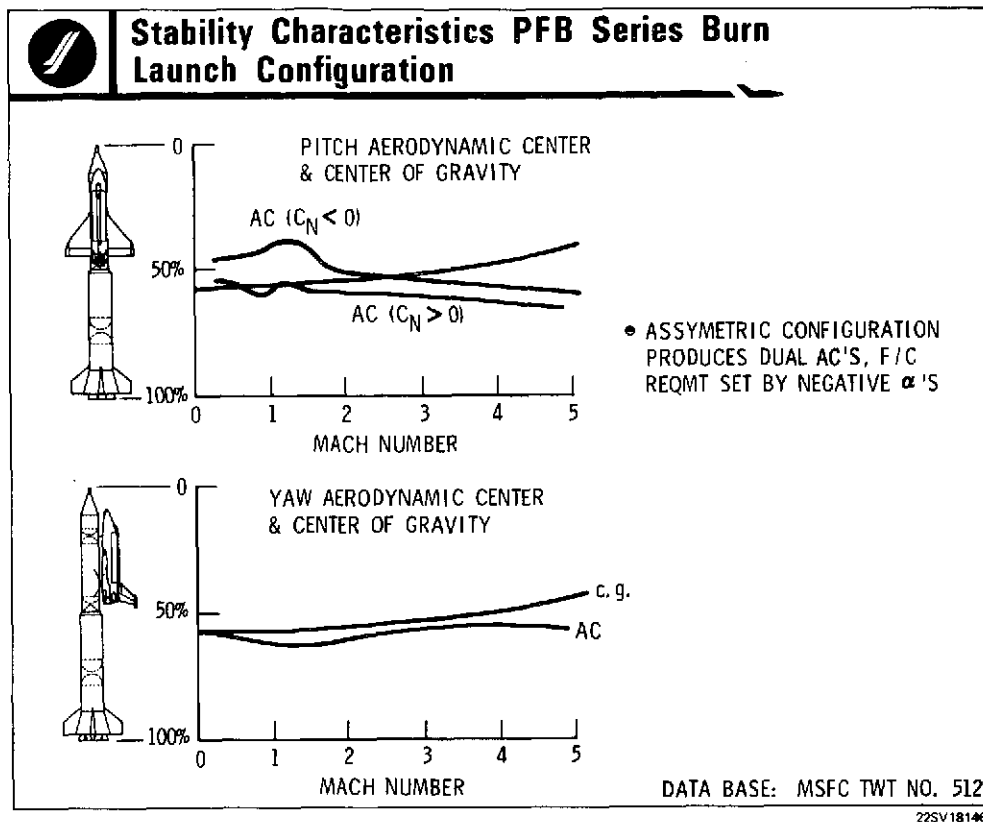
The aerodynamic characteristics shown are based on wind tunnel data from MSFC TWT test No. 512. The data have been adjusted to account for differences between the PFB baseline configuration and the model tested. The primary configuration changes considered in adjusting the test data were: an orbiter wing growth of 9.2 percent; a booster fin growth of 7.4 percent, and a 20-percent reduction to the orbiter vertical fin.

The two curves presented for pitch aerodynamic center reflect the dependence of booster fin effectiveness on launch vehicle attitude. As the angle of attack range is traversed from positive to negative angles, the interference effects of the orbiter on the booster fins are greatly increased for an angle of attack range from approximately -7 to -1 degrees. What results is a significant decrease in booster fin effectiveness. This loss in fin effectiveness is accompanied by a discrete shift of pitch aerodynamic center, making the vehicle less stable for the negative normal force condition. Therefore, negative angles of attack present the more critical condition in pitch static stability considerations.

The vehicle asymmetry which produces the dual pitch aerodynamic center condition does not exist in the yaw plane. Consequently, a single curve is presented for yaw aerodynamic center. The graph shows that yaw static stability is achieved across the entire Mach range.

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Stability Characteristics
120 in. SRM Parallel Launch Config
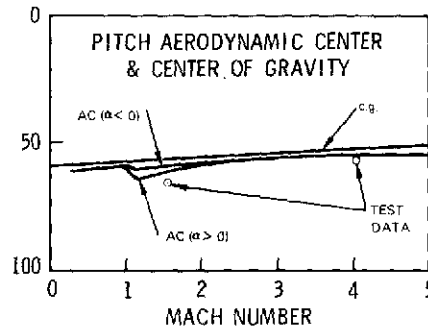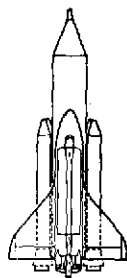
PITCH AERODYNAMIC CENTER & CENTER OF GRAVITY
AC (α< 0)
c.g.
AC (α> 0)
TEST DATA
MACH NUMBER

YAW AERODYNAMIC CENTER & CENTER OF GRAVITY
c.g.
AC
TEST DATA
MACH NUMBER

- WIND TUNNEL RESULTS CONFIRM ESTIMATES

- PITCH & YAW STABILITY OVER MACH RANGE

- ASYMMETRIC CONFIG · PRODUCES DUAL PITCH AC'S; F/C REQMT SET BY POSITIVE α'S
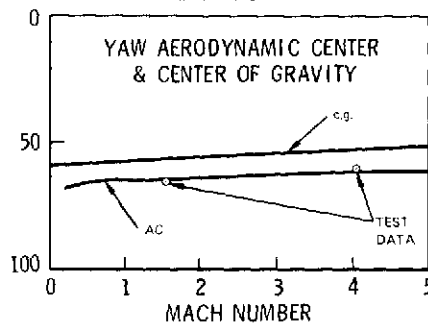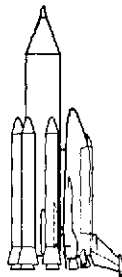
DATA BASE: GDHSWT TEST NO. 326-0

22SV180868

The aerodynamic center curves presented reflect NR's initial estimates of static stability for the 120-inch solid rocket motor parallel-burn launch configuration. These estimates were later substantiated by wind tunnel test data obtained in the General Dynamics wind tunnel (GDHSWT) Test No. 326-0. These test data also are shown on the graphs.

The vehicle asymmetry in the pitch plane produces two pitch aerodynamic center curves, one for positive angles of attack and another for negative angles of attack. This condition reflects the dependence of normal force slope distribution on vehicle pitch attitude. At negative angles of attack, the configuration exhibits less pitch static stability than for positive angles of attack. Consequently, pitch control requirements are established by the characteristics which exist in the positive angle of attack range.

Vehicle asymmetry does not exist in the yaw plane; consequently, yaw aerodynamic center is not dependent upon sideslip angle and only one aerodynamic center curve is shown. The vehicle exhibits yaw static stability across the entire Mach range. Since the configuration is more stable at positive angles of attack, flight control requirements are established in this range.

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PERCENT

PITCH AERODYNAMIC CENTER & CENTER OF GRAVITY

- WIND TUNNEL RESULTS CONFIRM ESTIMATES

- PITCH & YAW STABILITY OVER MACH RANGE

- ASYMMETRIC CONFIG PRODUCES DUAL PITCH AC'S - F/C REQMT SET BY POSITIVE α'S

YAW AERODYNAMIC CENTER & CENTER OF GRAVITY

DATA BASE: GDHSWT TEST NO. 326-0

22SV18085B
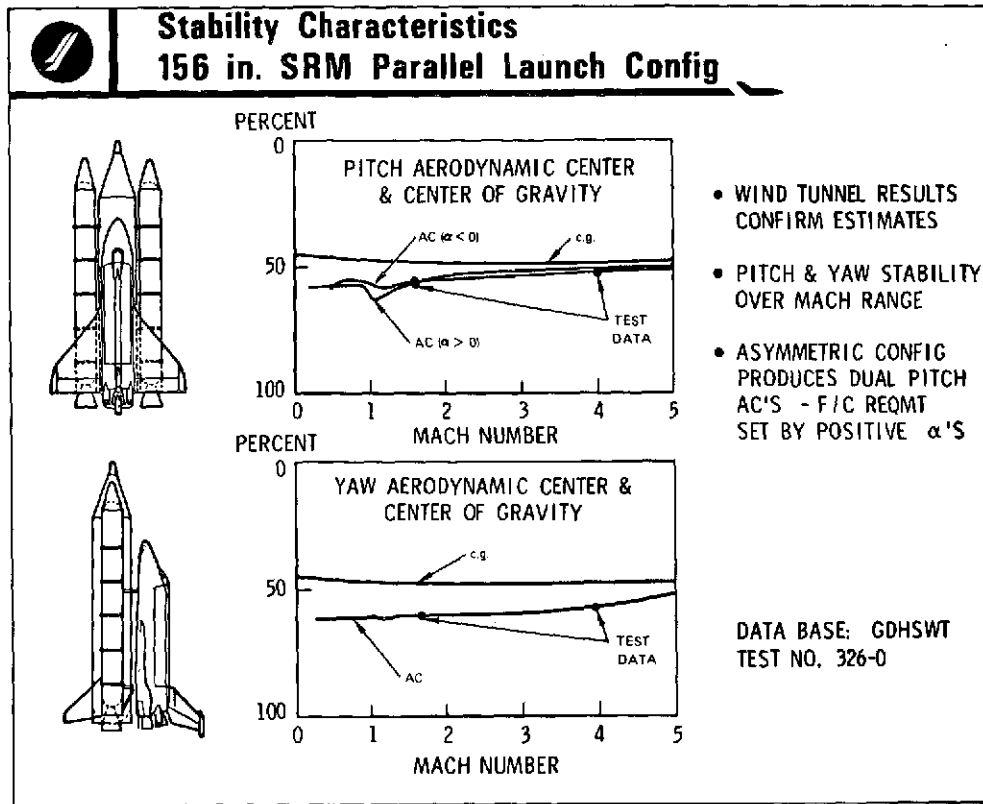
Wind tunnel results from GDHSWT test No. 326-0 are presented at Mach numbers 1.62 and 4.0 as substantiating evidence of pitch and yaw aerodynamic center predictions over the Mach range for the present 156-inch solid rocket motor configuration. It is evident from both the predicted data and the test results shown that static stability is achieved over the boost Mach range.

Vehicle asymmetry results in separate pitch AC curves for positive and negative angles of attack. Both the predicted data, which were based on modifications to MDAC Wind Tunnel Test No. S-222 results for a similar configuration, and the substantiating data points from the GDHSWT test exhibited considerable nonlinearity in the pitching moment and normal force coefficients in the region near zero angle of attack. However, it was possible to obtain a reasonable representation of the test data for both configurations by considering two linear ranges, one for $\alpha > 0$ and the other for $\alpha < 0$. Only one yaw AC curve is required due to configuration symmetry about the X-Z plane.

Rolling Moment Due to Side-Slip

Roll Control Best for PFB Configuration

Ventral Fins Reduces Rolling Moment Approximately 30%

120 in. SRM Config; Data Points (□)

156 in. SRM Config (W/O Ventral Fins); Data Points (▽)

156 in. SRM Config (With Ventral Fins)

PFB Config

Data Base: GDHSWT
Test No: 326-0
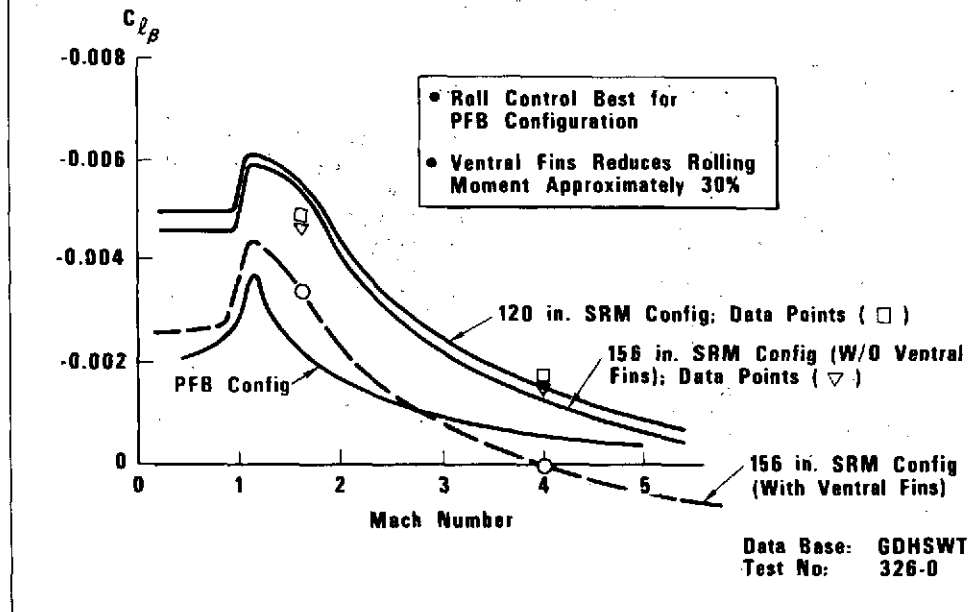
22SV18227B

The curves presented in this chart show the rolling moment characteristics due to sideslip for the 120-inch and 156-inch parallel-burn solid rocket motor and series-burn pressure-fed-booster configurations. The pressure-fed booster configuration has greatly reduced rolling moment because of the increased effectiveness of the lower booster vertical fin and lower portion of the booster flared skirt. Washout from the orbiter minimizes the effectiveness of the upper portion of the flared skirt and upper fin.

Test results from the General Dynamics wind tunnel test No. 326-0 are presented at Mach numbers 1.62 and 4.0 to substantiate the data predictions. The reduction in rolling moment due to the addition of ventral fins to the 156-inch solid rocket motor boosters also is illustrated, making it comparable with the pressure-fed booster configuration.

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0.00465 Scale
GD Wind Tunnel
Mach 1.6 & 4.0
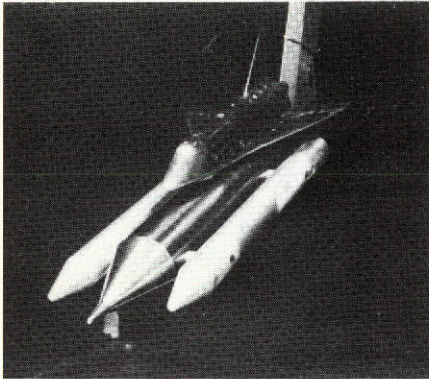
Launch Config
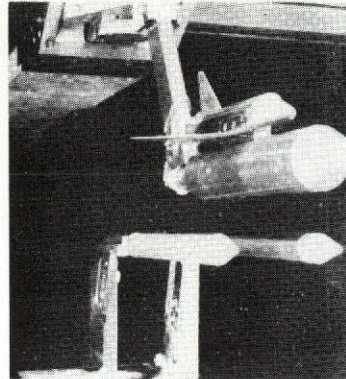
Separation

22SV18074

The photograph shows a wind tunnel model utilized for both force and separation tests in the General Dynamics wind tunnel. Data were taken at Mach 1.6 and 4.0 on a 0.00465-scale model.

**0.00465 Scale
GD Wind Tunnel
Mach 1.6 & 4.0**

22SV18075
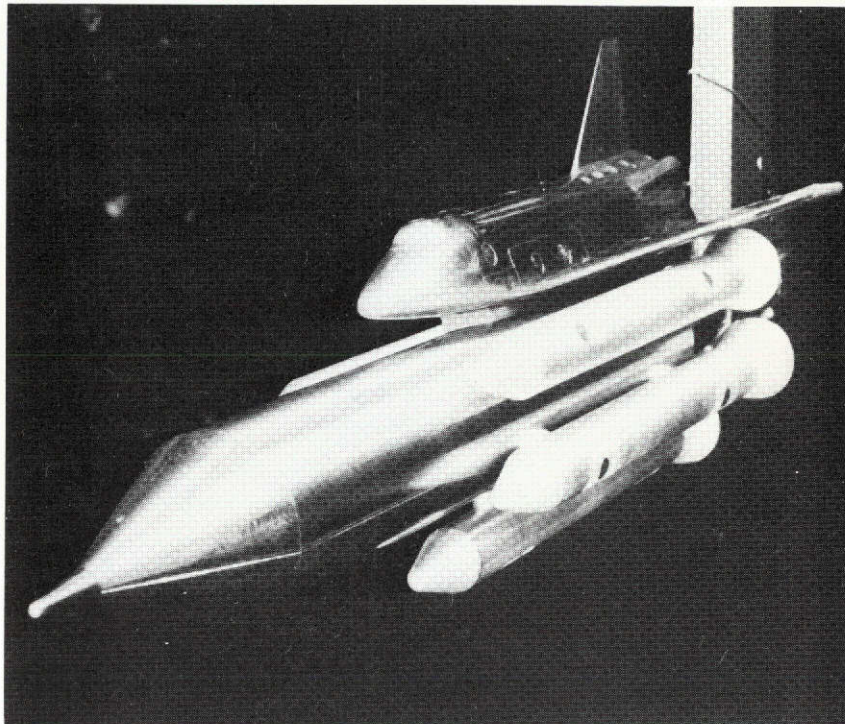
The photograph shows the testing of a 0.00465-scale wind tunnel model utilized to obtain force data at Mach 1.6 and 4.0 in the General Dynamics wind tunnel. These data and the 156-inch solid rocket motor wind tunnel test data were used in flight control and separation studies.

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FLIGHT CONTROL

Flight Control Evaluation

	Alternate Configurations	Control Options
Series	PFB	Booster TVC
Parallel	120 In. SRM	Orbiter TVC
	156 In. SRM	Aerosurfaces
		Fin

Issue:
- Is Booster TVC Required

Requirements:
- Load $q\beta = 2400$ $q\alpha = 2800$
- Roll Control $< 30°$ With Winds
- Minimize Sum of Structural and Propellant Weight

Considerations:
- c.g. Travel
- Winds/Gusts
- Thrust Misalign/Mismatch
- Orb Engine Out
- Actuator Failure

22SV17927B

The key issues in the flight control studies were to verify the control
requirements for the pressure-fed series-burn system and to deter-
mine the most appropriate control mode for the parallel-burn systems,
and in particular to determine whether or not thrust vector control was
required on the parallel-burn booster motors. In these analyses load
limits as illustrated on the chart and roll limits with winds were
imposed on the system. Tradeoffs were made to minimize the sum of
the structural penalty and the additional propellant weight required to
compensate for dispersions. The analyses considered the require-
ments to track c.g., to control through winds and gusts, to compen-
sate for thrust misalignments and thrust level mismatches, to provide
adequate control with one orbiter engine out or with one actuator
failure on any system. The chart illustrates the considerations in
analyzing the control requirements for the pressure-fed booster.

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Tight Roll Control

Tight Roll Control

NO WINDS	344	2800
HEAD WINDS	-6180	
WIDE WINDS	1620	
TAIL WINDS	4360	

$\bar{q}\alpha$ - PSF-DEG X 10^{-3}

	51	2400
	56	
	5634	7° ROLL
	50	

$\bar{q}\beta$ - PSF-DEG X 10^{-3}

	Y	P
	1.4	
	2.4	
	(4.3)**	5.0
	1.7	

δ MAX, DEG

* LOOSE ROLL CONTROL
** Avg of All Engines at Wind Spike
*** Increases to 4140 PSF-Deg with 43° Roll for Eng Out

32K Ft Gust, Pure Attitude

		2800
TAIL WINDS	3257	
HEAD WINDS	-3412	
SIDE WINDS	-625	
SIDE* WINDS	1888	
HEAD QTRING WINDS	-2541	
TAIL QTRING WINDS	1715	

$\bar{q}\alpha$ - PSF-DEG X 10^{-3}

		2400
	107	
	134	
	*** 3789, 12° ROLL	
	2826, 57° ROLL	
	2583, 7° ROLL	
	3097, 7° ROLL	

$\bar{q}\beta$ - PSF-DEG X 10^{-3}

	Y	P
	2.9	
	3.0	
	(0.8)**	5.0
	(1.6)**	5.0
	(2.4)**	4.2
	(P = 1.4)** (Y = 1.6)**	5.0

δ MAX, DEG

32K Ft Gust, Load Relief

22SV18218A

The chart displays the aerodynamic load parameters and engine thrust vector control requirements for the pressure-fed booster for two modes of control where the roll attitude is closely restricted. It is seen that using pure attitude control flying through a gust at 32,000-foot altitude that extremely high $\bar{q}\beta$ loads are incurred due to side winds, although the TVC requirement is within the five-degree capability of the engines. With a load relief system and allowing up to 12 degrees roll, a $\bar{q}\beta$ of 3789 PSF-degrees is incurred. It is believed that with further study this value can be reduced together with other values which exceed the $\bar{q}\alpha$ and $\bar{q}\beta$ limits to acceptable values. Again in this system the five-degree limit on thrust vector control is not exceeded.

The TVC values shown on the chart include single-engine peak values as well as the average value of all engines at the time of the peak requirement.

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Control Reqmts 5-120 in. SRM Parallel Burn

- Analyzed at Lift-Off & SRM Burnout
- Low q̄ -Aero Surfaces Ineffective
- Considered Engine, Actuator Failure, Thrust Misalignments, c.g. Travel

? Is Vehicle Control Using Orbiter TVC Possible?

39×10 Ft-Lb

5'

B/O

L/D

c.g. 7°

23°

16°

Thrust Moment

$T_{resultant}$

Use Booster TVC
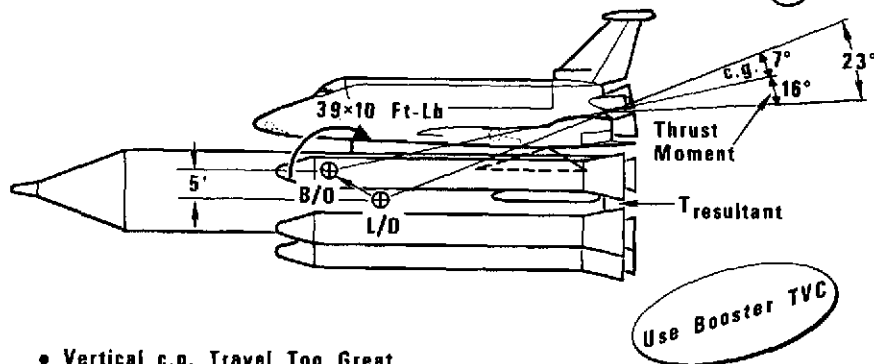
- Vertical c.g. Travel Too Great
- Cannot Reposition 5 SRM for Satisfactory Result

22SV17929A

An examination was made to determine whether the orbiter main propulsion system could track the total system center of gravity from liftoff to solid rocket motor burn out and at the same compensate for engine failure, actuator failure, and thrust misalignments. It was determined that the total requirement was well in excess of the engine capability (±10 degrees). In addition, the vertical c.g. travel of the configuration was too great. No repositioning of the five motors provides a satisfactory compromise. Thus, it was concluded that booster thrust vector control is a requirement.

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Alternate 120 in. SRM Configuration

•* ENGINE & ACTUATOR FAILURES   • ** THRUST MOMENT DUE TO
   THRUST MISALIGNMENTS   c.g. TRAVEL
   c.g. TRAVEL
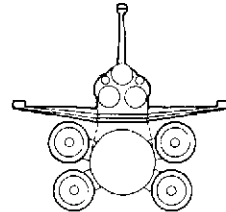
CONFIGURATION	ORBITER GIMBAL ANGLE REQUIREMENTS		
	δ^*	δ^{**}	δ TOTAL
4-120 IN.	14^0	3.5	17.5

22SV1B147B

An alternate configuration with 120-inch SRM motors was examined where
some decrease in vehicle capability was accepted. This system included
four 120-inch SRM's located as shown. Because of the substantial reduction
in c. g. travel, the total orbiter gimbal angle requirements were reduced
to a value within the engine capability. It was then desired to examine how
best to complement the orbiter TVC to provide the necessary steering and
disturbance control capability. This is described in the next chart.

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120 in. SRM Roll/Yaw Control

Pitch
- 5 SRMS - Orbiter TVC Inadequate
- 4 SRMS - Orbiter TVC Marginal

Roll
- Orbiter TVC Insufficient Authority

Orbiter TVC & Aero Surface Control
- Increased Hinge Moments × 3
- Hydraulic System Impact
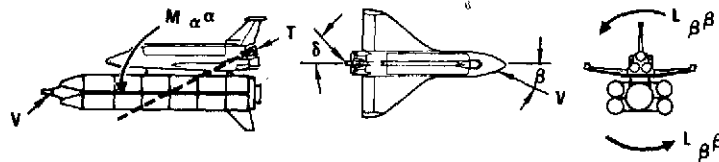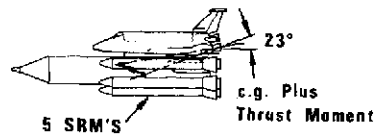
Orbiter TVC & Aero Surface Control & Ventral Fins
- Separation Problems
- Hydraulic System Impact

Booster TVC
- Acceptable Control

Booster "Trim" TVC & Orbiter TVC & Aero Surface Control
- Hydraulic System Impact

5 SRM'S c.g. Plus Thrust Moment 23°
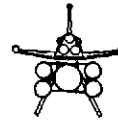
22SV18200B

Dynamic analyses immediately illustrated that the orbiter thrust vector control was not capable of controlling roll. Supplementing the orbiter thrust vector control with orbiter aerosurface control within the current hinge moment limitations resulted in a roll displacement of greater than 100 degrees and rates up to 20 degrees per second. To decrease the roll displacement and rate to a reasonable value, the hinge moment increased by at least a factor of 3.

An alternate concept involving the use of ventral fins indicates that acceptable control is possible with the current hinge moments. However, there was a significant weight impact with the addition of the fins, an impact on the hydraulic system because of the requirement to actuate the aerodynamic surfaces and the orbiter TVC simultaneously, and finally an added complexity to the separation problem because of the fins.

The option wherein the only mode of control would be booster TVC was determines to be an acceptable control system without orbiter impact.

A last option investigated included the use of booster TVC to provide trim only. The orbiter thrust vector control and aerosurfaces would provide control for disturbances. In this system, a simple blowdown hydraulic system was assumed. Again because of the use of orbiter thrust vector control and aerosurface control simultaneously, there would be an impact on the orbiter hydraulic system. This last system would be somewhat more involved and risky than the booster thrust control vector only system although perhaps somewhat less costly. This particular trade study was performed on the 156-inch SRM parallel-burn system but the results are felt to also be applicable to the 120-inch solid rocket motors. Therefore, thrust vector control on the booster only is recommended for this system.

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12.6K Winds

PURE ATTITUDE

	$\bar{q}\alpha$, PSF-DEG X 10^{-3}	$\bar{q}\beta$, PSF-DEG	W$_S$ - LB X 10^{-3}	δ MAX, DEG
NO WIND	1308 (2800)	69 (2400)	406	3.6
HEAD WIND	-3747	83	5367	3.6
SIDE WIND	2040	4600	408	(4)* 5.0
TAIL WIND	4727	58	800	2.5

$\phi_{MAX} = 27°$

LOAD RELIEF

	$\bar{q}\alpha$, PSF-DEG X 10^{-3}	$\bar{q}\beta$, PSF-DEG X 10^{-3}	W$_S$ - LB X 10^{-3}	δ MAX, DEG
NO WIND	1224 (2800)	61 (2400)	403	3.6
HEAD WIND	-2204	60	2272	3.7
SIDE WIND	1371	3187	419	(1)* 5.0
TAIL WIND	3371	98	643	3.6

$\phi_{MAX} = 30°$
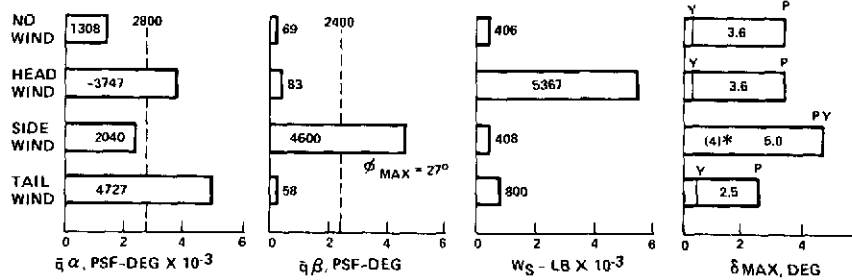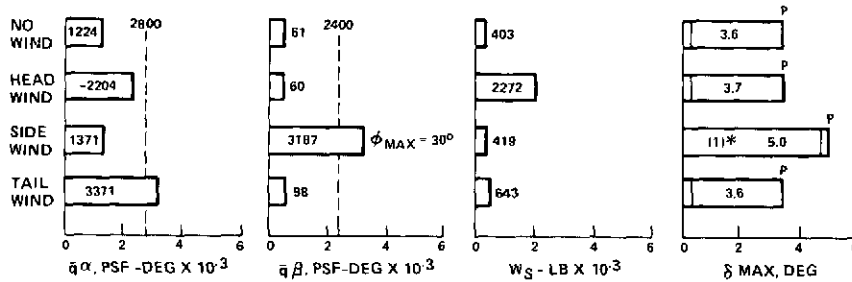
22SV17928B

The data resulting from a dynamic analysis of the control requirements for a five-engine solid rocket motor system utilizing 120-inch solids with booster thrust vector control only illustrate that with load relief the $\bar{q}\alpha$ and $\bar{q}\beta$ limits can be approached. Again, fine tuning of this system will reduce those values which exceed the design requirements. Good roll control is obtained under most circumstances. A booster nozzle deflection limit of five degrees is adequate.

Control Reqmts
2-156 in. SRM Parallel Burn

● * Engine & Actuator Failures
   Thrust Misalignments
   c.g. Travel

● * * Thrust Moment
    Due to c.g. Travel

Configuration	$\Delta Z_{c.g.}$	$\delta *$	$\delta * *$	δ Total
Symmetrical	34°	9°	18°	27°
Skewed	0	9°	0	9°

78 in.
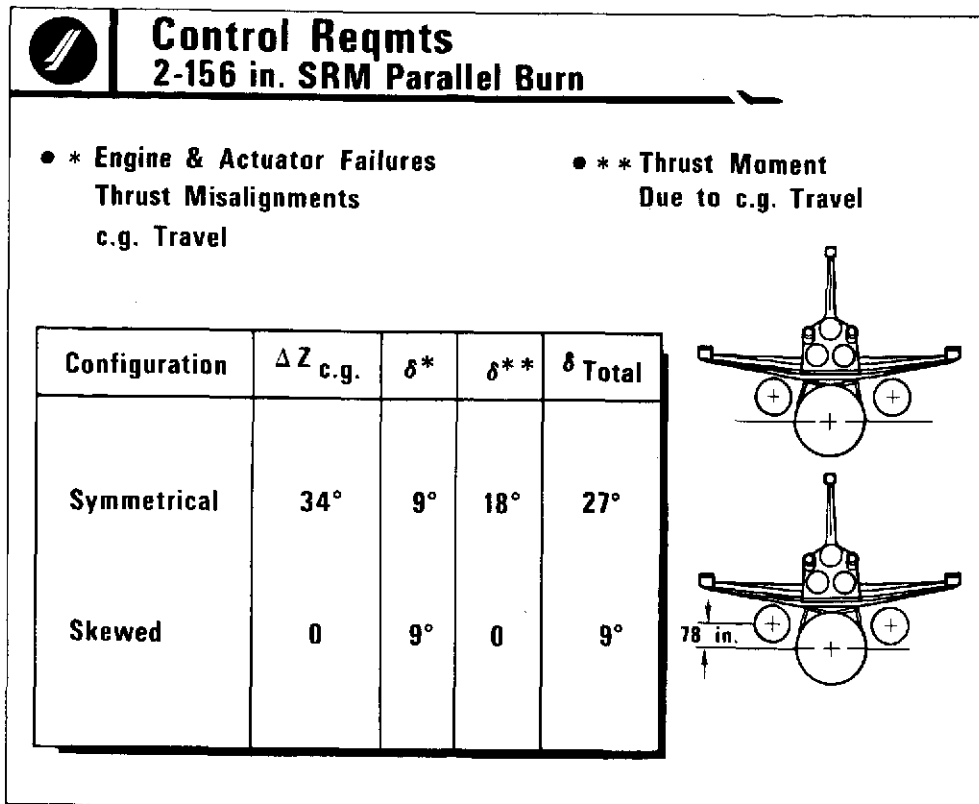
22SV17930A

The requirements to track the c. g. and compensate for engine actuator failures and thrust misalignments were examined for two configurations. In one, the booster engines were located on the external oxygen-hydrogen tank centerline, and in the second, the booster engines were located 78 inches above the booster centerline. This latter configuration reduced the orbiter thrust vector control requirements for c. g. tracking and engine or actuator failures to only nine degrees. The apparent difficulty of separating these motors resulted in the elimination of this configuration. It is clear then that the control capability of the orbiter engines must be supplemented to provide adequate control.

156 in. SRM Roll/Yaw Control

Pitch
- Symmetrical - Orbiter TVC Inadequate
- Skewed - Orbiter TVC Adequate but Severe Separation Problems

Roll
- Orbiter TVC Insufficient Authority

Orbiter TVC & Aero Surface Control
- Increased Hinge Moments × 3
- Hydraulic System Impact

Symmetrical (Large c.g. Excursion)   78 in.   Skewed
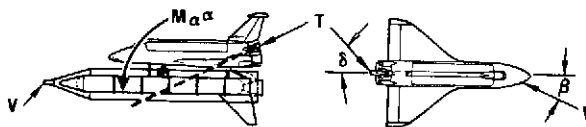
Orbiter TVC & Aero Surface Control & Ventral Fins
- Weight Impact, (2 × 650 Ft² Fins)
- Hydraulic System Impact
- Separation Problems

Booster TVC
- Accceptable Control

Booster "Trim" TVC & Orbiter TVC & Aero Surface Control
- Hydraulic System Impact
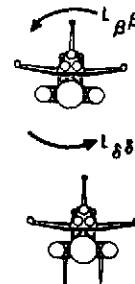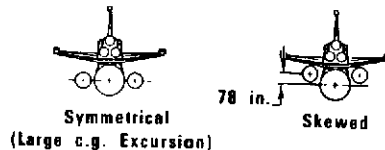
22SV17926D

Initial dynamic analyses showed that the orbiter TVC had insufficient roll authority. A second option wherein aerosurface control was used to supplement the orbiter TVC, resulted in extremely high hinge moments if the roll displacement and rate were to be controlled. In all, a significant hydraulic system impact was seen. Another option wherein ventral fins were used to supplement the orbiter TVC and aerosurface control to trim the roll moment resulted in an acceptable control situation. However, a significant weight impact from the addition of the fins and the growth of the hydraulic system resulted. Again, added separation problems due to the presence of the ventral fins would be anticipated.

The use of booster thrust vector control provided acceptable control in all regards.

A final option wherein the booster motors were used for trim control only and the orbiter TVC and aerosurface were utilized for control of aerodynamic disturbances resulted in an impact to the hydraulic system because of the parallel utilization of the orbiter TVC and aerosurface control.

Booster Control Option Trades-156 In. SRM

	Weight Δ (Lb)			Program Cost Δ ($ × 10^{-6})			
	Orbiter	Booster	Fins	Orbiter	Booster	Fins	Total
Booster TVC δ = ±5°, δ̇ = 5°/Sec		16,000			116		116 ✓
Orbiter TVC + Aerosurface + Fins (Nominal Hinge Moments)	1,690		8,000	28		296	324
Orbiter TVC + Aerosurfaces (3 × Hinge Moments)	14,000			225			225
Orbiter TVC + Aerosurfaces + "Slow" Booster Trim in Pitch δ = ±3°, δ̇ = 0.1 Deg/Sec	1,690	1,200		28	58		86 *

*Complex Control Blending

22SV18121B

A trade study was completed to determine the program cost impact
of various modes of control for the 156-inch solid rocket motor
parallel-burn system. These are illustrated in the figure. It is
seen that the lowest cost system in terms of program impact is
the orbiter thrust vector control plus aerosurfaces for disturbance
control with a slow booster trim in pitch. The program impact
of this configuration was 86 million dollars. However, because
of the relative complexity associated with blending the three
different control modes, it was recommended that for the
parallel-burn systems (156-inch as well as 120-inch) that booster
TVC only with deflections up to five degrees at rates of five
degrees per second can be utilized.

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Control Summary

- SERIES PFB - ±5° TVC

- PARALLEL 5 X 1207 SRM - ±5° SRM TVC REQUIRED

- PARALLEL $\left\{\begin{array}{l} 4 \text{ X } 1207 \text{ SRM} \\ 2 \text{ X } 156 \text{ SRM} \end{array}\right\}$ - $\begin{array}{l} \text{±5° SRM TVC} \\ \text{OR} \\ \text{ORBITER MPS TVC} \\ \text{ORBITER AILERONS} \\ \text{VENTRAL TRIM} \\ \text{SURFACES} \end{array}\left.\begin{array}{l}\\\\\\\end{array}\right\} \begin{array}{l} \text{HYDRAULIC} \\ \text{POWER} \\ \text{INCREASE} \end{array}$

22SV182058

The parallel burn systems incorporating two 156 solid rocket motors
(SRM's) or four 120 SRM's can be controlled with 5 degrees SRM
thrust vector control (TVC) or orbiter TVC with ailerons and ventral
trim surfaces. For the parallel burn systems, it is recommended
that the minimum risk system with good cost effectiveness would be
booster TVC only. The result is that booster TVC is required on
all systems and therefore is not a selection discriminator.

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Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
• SRM TVC Requirements	• Required for All Systems
• Stage Separation	
• SRM Thrust Termination Requirements	
ABORT	
• Mode	
• Control	
• MPS Requirements	
ENVIRONMENT IMPACT	
TEST & FACILITY IMPACT	

22SV18243B

Separation analyses were made for the series burn pressure-fed system and for the two parallel burn systems. Because previous analyses have been completed for the series burn system, the effort concentrated on the parallel systems.

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Tank Booster Separation System

Interim Design Apollo MDF Concept
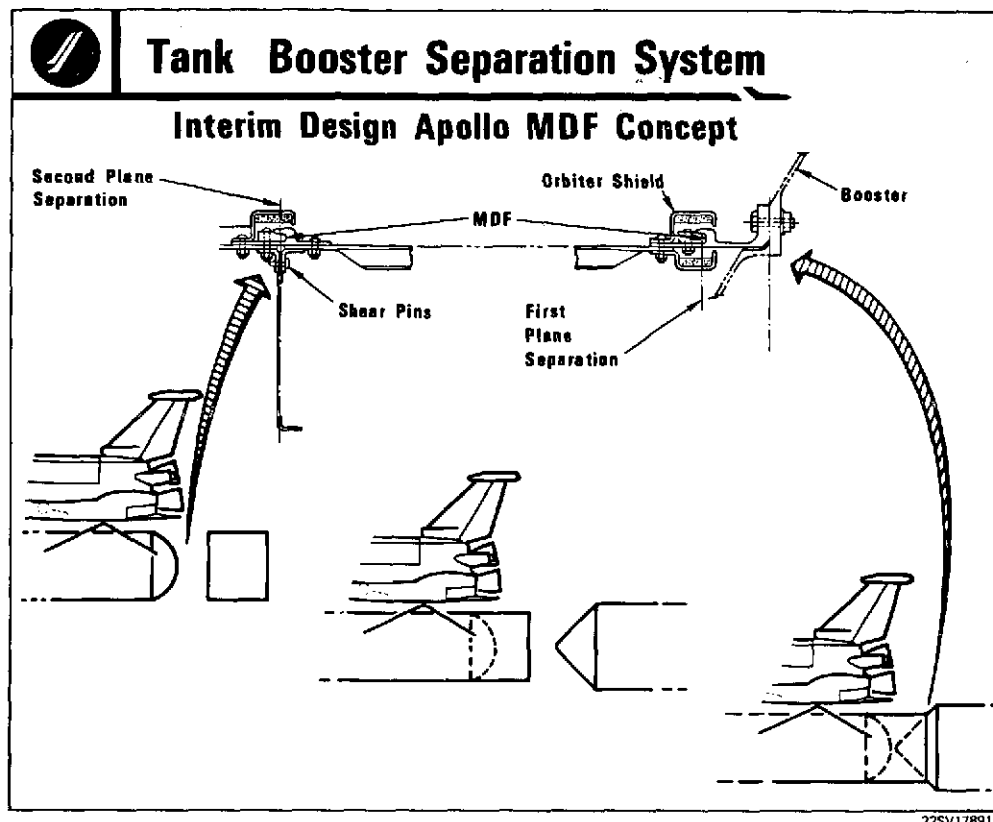
A preliminary design of the separation plane systems for the pressure-fed booster (PFB) series burn separation concept is shown. Both the first plane and second plane separations are accomplished through the use of a mild detonating fuse (MDF) used to cut through the circumferential structure. Shields are provided to retain separation system fragments.

1207 SRM Separation with Translating Linkage
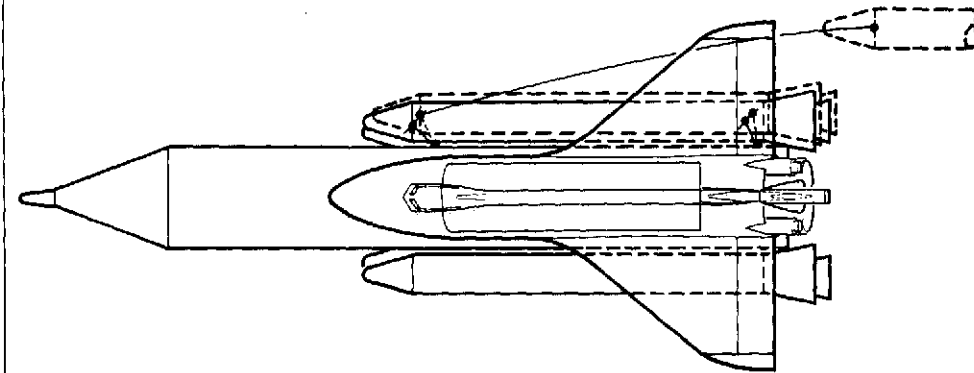
22SV17890B

The concept studied for separating the 1207 solid rocket motors as well as the 156-inch solid rocket motors is illustrated. In this concept, hinged links fore and aft which provide separation as the links go into tension are used to assure positive displacement of the solid motors. The length of the hinges are adjusted to provide positive separation and simultaneous release of all links from the motors. The same system is used for the motor mounted in the pitch plane and those mounted on the sides of the external oxygen-hydrogen tank (EOHT).

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156 in. SRM Mating & Separation Sys

ORBITER TANK

SRM

THRUST LINK

TANK

SRM

EXPANSION CYLINDER

LATERAL LINK

LATERAL LINKS RELEASED

THRUST LINK RELEASED

LATERAL LINK
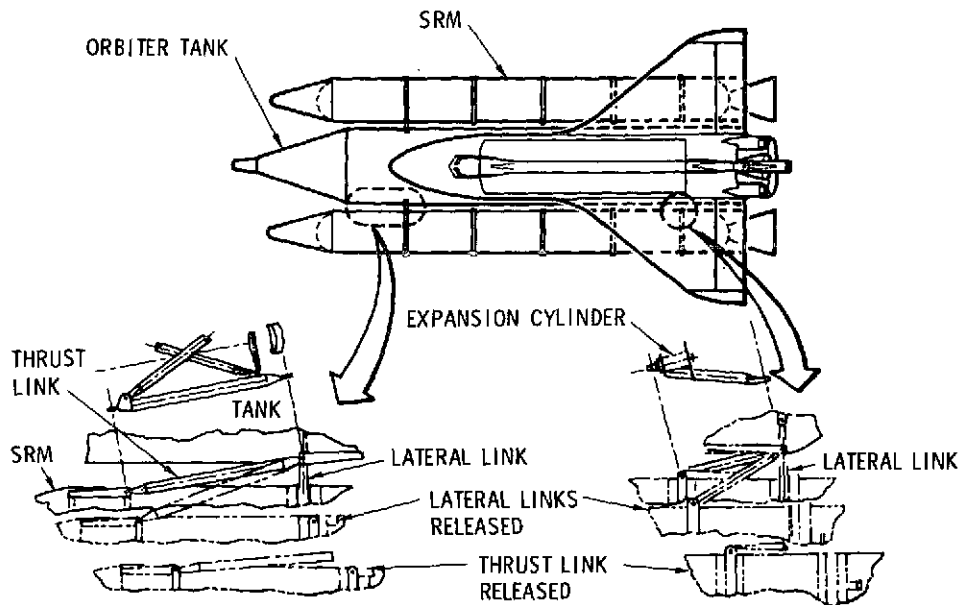
22SV18114

Details of the linkage system are provided in the opposite chart. Of key significance is the expansion cylinder designed to provide for expansion of the rocket motors under thrust and the contraction of the external oxygen-hydrogen tank at cryogenic temperatures. The linkage system provides for transmission of booster thrust as well as booster support and separation. The thrust is taken through the forward linkage system.

PFB Orbiter-Tank/Booster Separation

Concept-Dual Plane Separation

Nominal Staging

$\overline{q}$ = 50 PSF
Alpha = 0 Deg

Interstage 1ST Separation Plane
Note: 2ND Separation Plane is
Initiated at T = 10 Sec

PF Booster

T = 2.00 Sec
T = 1.40 Sec
T = 0.00 Sec

Orbiter

EOHT
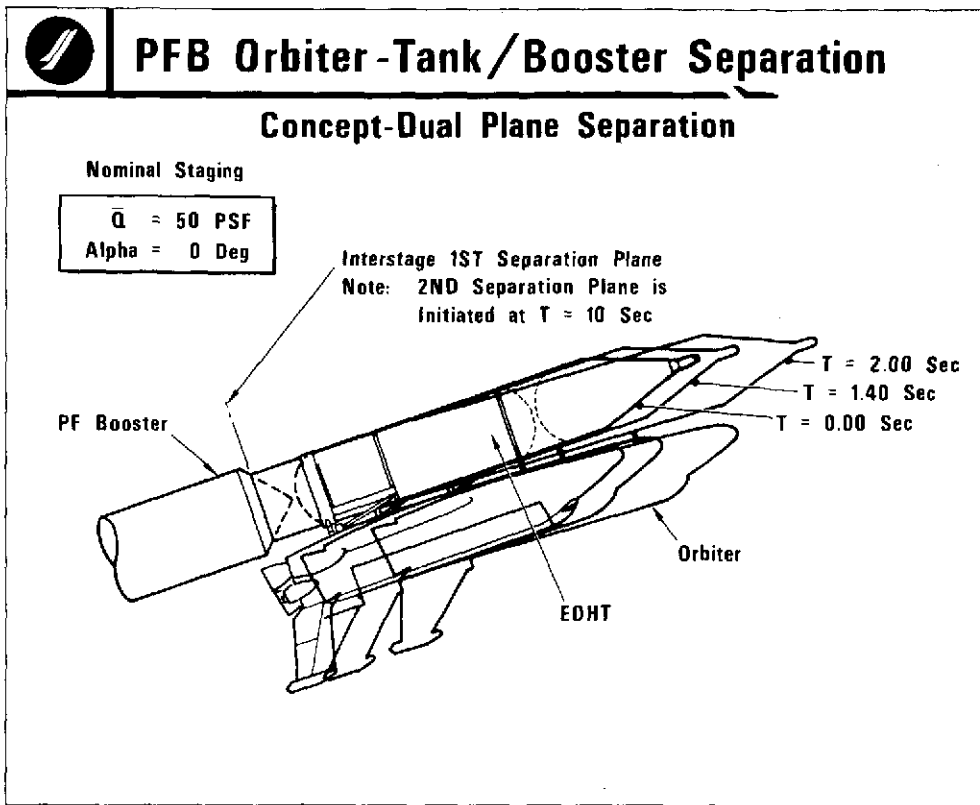
22SV18101B

This chart describes the separation dynamics of the pressure-fed
booster orbiter system from the booster at nominal staging. It is seen
that the orbiter and external oxygen-hydrogen tank are completely
clear of the booster at two seconds. Additional analyses are being
conducted at high $\overline{q}$ and at angles of attack to verify separation com-
pletion under abort situations.

120 in. SRM Orbiter-Tank/SRM Separation

Concept-Hinged Links

MAX $\bar{Q}$ SEPARATION

$\bar{q}$ = 650 PSF
α = 0

T = 0.0 Sec
T = 1.5 Sec
T = 2.0 Sec
T = 2.5 Sec

NOMINAL STAGING

$\bar{q}$ = 56 PSF
α = 0

T = 0.0 Sec
T = 1.5 Sec
T = 2.0 Sec
T = 2.5 Sec
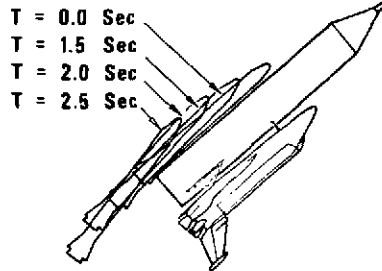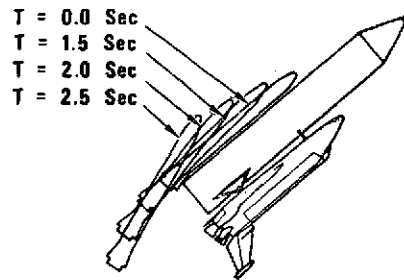
Propulsion Characteristics at Separation

- Orb MPS: Full Thrust
- 120 in. SRM: 54K Each (Abort)
- Abort SRM: 134.5K Each

22SV18081C

The trajectory of a pitch plane 120-inch solid motor during separation sequences at max $\bar{q}$ and at nominal staging is shown. The orbiter main propulsion system is brought up to full thrust before starting separation.

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Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
• SRM TVC Requirements	• Required for All Systems
• Stage Separation	• Series Burn - State-Of-The-Art Parallel Burn - Intensive Effort Required
• SRM Thrust Termination Requirements	• Required for Abort on All Systems
ABORT	
• Mode	
• Control	
• MPS Requirements	
ENVIRONMENT IMPACT	
TEST & FACILITY IMPACT	

22SV18244B

It is seen that the separation concept for the series burn is state of
the art and that no significant problems are anticipated with this
system. For the parallel burn system, intensive effort is required
to develop, test, and verify the adequacy of the concept. Thrust
termination ports also are required on solid motors to be used in all
abort situations.

Mission Termination Requirements

- Phase B' Requirements
  - Provide for Rapid Egress of Crew & Passengers Prior to Liftoff
  - Provide Intact Abort Capability for all Flight Phases (Goal)

- Criteria
  - Prelaunch

    Time for Crew/Passenger Egress to Safe Area ~ 60 Sec
    Pad Flyaway

  - Post Launch

    Provide for Safe Recovery of Individual Stages

    Crew/Passengers Safety Prime Consideration
    (Hardware Loss or Damage Secondary)

    Provide for Land Recovery of Orbiter

    Provide Separation Capability for All Flight Regimes

22SV17957A

During the Phase B'' study, requirements were established to provide for rapid egress of crew and passengers before liftoff. Trade studies were completed to determine the most appropriate mode of egress and transportation of the crew and passengers to a safe area. In addition, intact abort capability for the crew, passengers, and orbiter for all flight phases was to be provided.

During the pre-launch sequence, it was a requirement that the crew and passengers could attain egress to a safe area within 60 seconds. For the orbiter, it was a requirement that the orbiter could separate safely from the booster and attain an intact landing any time after commit to launch.

Launch Pad Emergency Egress

Alternates

- Emergency Egress Options
    1. Hi-Speed Elevators & Slide Wire (Apollo/Sat V Sys)
    2. Semi-Free Fall Elevators
    3. Free Fall Device

Timeline Free Fall (SRM & PFB) 6 Man Unaided Escape

```
0        10        20        30        40        50        60
         V = 10 Ft /Sec
```

First Man to Access Arm ⎦

First Man to Enter Cage ⎦

4th Man Enter Cage ⎦

6th Man Enter Cage & Strap Down ⎦

Head Count & Release ⎦

Slide Wire (Backup)

PFB - 150 Sec

SRM - 126 Sec

Complete Descent ⎦

Enter Blast Shelter ⎦
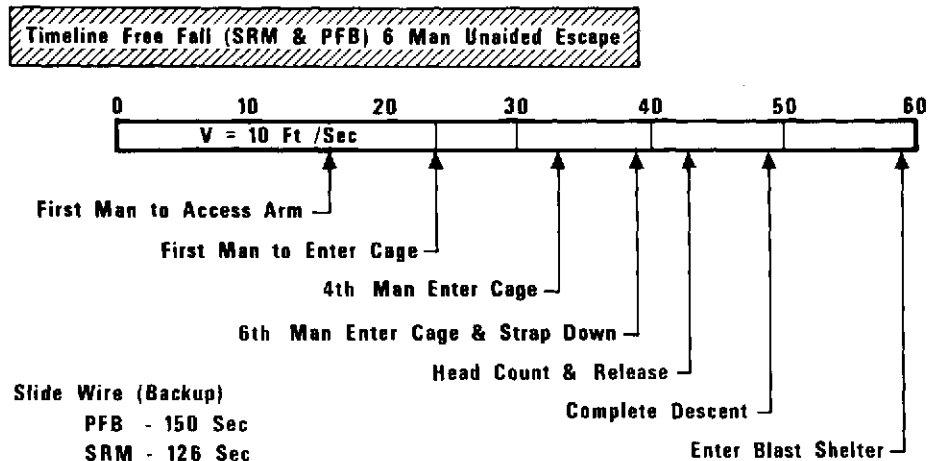
22SV18047C

To provide for emergency egress for the crew and passengers, three options were examined: high-speed elevators and slide wire similar to those used on the Apollo/Saturn V system, semi-free-fall elevators, or a free-fall device. It was determined that the free-fall device could meet the limitations on egress to a safe area as well as provide a cost-effective solution. The chart illustrates the timeline from initiation of egress to entrance to the blast shelter at approximately 60 seconds.

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Vehicle Pad Abort Options

ASRM Wt - 94,210 Lb
F - 280K Each (2)
I_B - 37 Sec
Δ Cost - $293M

ASRM Wt - 54,324 Lb
F - 516K Each (4)
I_B - 5.0 Sec
Δ Cost - $299M

ASRM Wt - 94,210 Lb
F - 280K Each (2)
I_B - 37 Sec
Δ Cost - $293M

ASRM Wt - 94,210 Lb
F - 280K Each (2)
I_B - 37 Sec
Δ Cost - $293M

ASRM

ASRM

ASRM

ASRM

PFB

PFB

SRM 156 in.

SRM 120 in.

Orbiter Alone Abort

Orbiter Plus Tank Abort

Orbiter Alone Abort
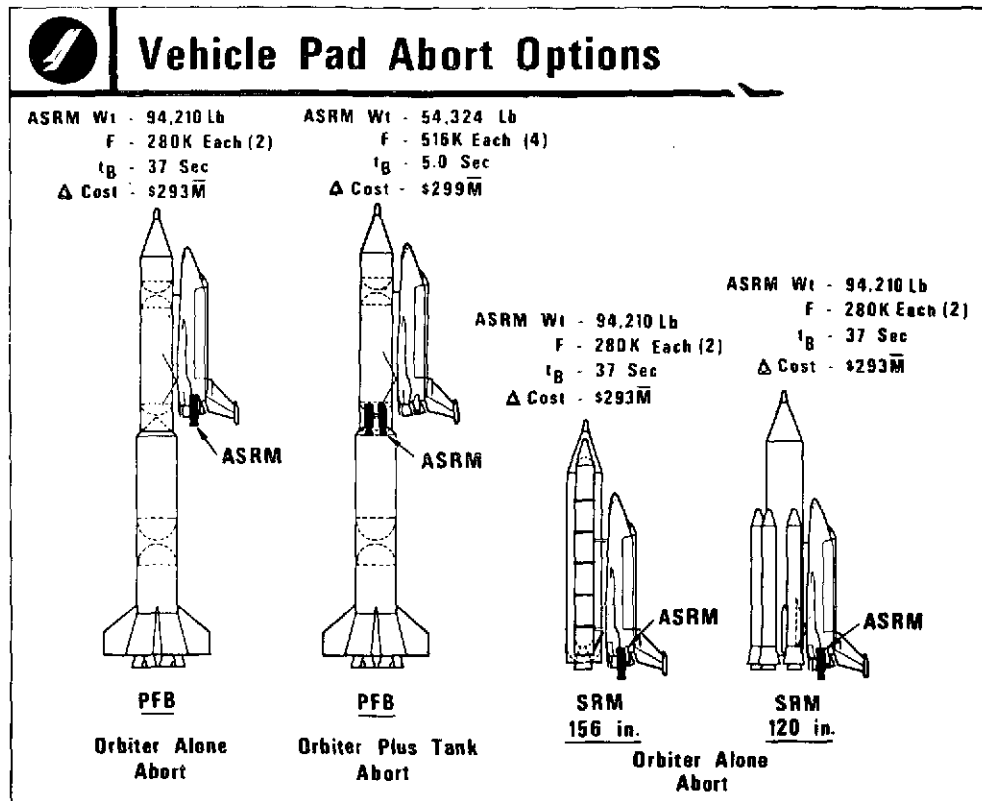
22SV1790DE

Two options were available to provide for safe abort of the orbiter. In one case, abort solid-rocket motors are mounted on the portion of the orbiter fuselage as illustrated in the chart. This solution is appropriate for either a series-burn configuration or for the parallel-burn configurations. A second option incorporates abort solid rocket motors on the interstage between the external oxygen-hydrogen tank (EOHT) and a series-burn booster such as the pressure-fed booster. This option is only appropriate for a series-burn system because both the orbiter and the EOHT abort from the booster. In a parallel system it would not be feasible to fly the orbiter and EOHT out from between the cluster of solid-rocket motors and the size of the EOHT would require extremely large abort solid rocket motors (ASRM's). It has been determined that the weight penalty imposed by the ASRM system is offset by the performance gained through the use of the abort system during the nominal mission. Specifically, the abort motors are ignited after nominal staging and fire in parallel with the orbiter motors. Careful sequencing of the ASRM firing is required together with possible throttling of the orbiter main propulsion system (MPS) to avoid overacceleration of the orbiter.

Cost estimates for these abort systems have been developed. In either case the total system cost would be approximately 300 million dollars.

- 40 -

Time	ASRM's on Orbiter	ASRM's on Interstage
Pad 99% Wind ±1/4° Thrust Misalignment	• Rotate Launch Vehicle on Pad (PFB) • Precant Launch Vehicle for Clearance 2.5 Deg • ±3° TVC on ASRM for Initial Control • Aero Surface Control Adequate After 15 Sec	• Precant Thrust 2 Deg • SSME TVC After 4 Sec
Max Q	T < 80 Sec: Abort Orbiter Only. Control with Aero Surfaces T > 80 Sec: Abort Orbiter & EOHT, Realign ASRM Thrust Through c.g. Control with SSME TVC	• Aero Stable - Ignite ASRM's for Separation, SSME TVC for Control
Staging	• Realign Thrust Vector Thru C.G. • Baseline Dual Plane Separation • Drop Interstage at 10 Sec • SRM Burn at 12 Sec After Separation	• Ignite ASRM's at Staging • Baseline Dual Plane Separation Sequence

22SV17923C

Each of the two options described previously impose control requirements on the vehicle and on the abort solid rocket motor (ASRM) system. The chart illustrates the control requirement for each of the most significant time sequences, that is, separation off the pad, max $\bar{q}$ and at nominal staging. The most significant difference between the two options is that in option 1, the ASRM's on the orbiter require thrust vector control (TVC) for the ASRM's during the off-the-pad launch. After 80 seconds of flight, the orbiter cannot return to the launch pad and must be separated with the EOHT. Because of the difference of location with the configuration center of gravity, the ASRM's must be repositioned to permit the thrust vector to pass through the configuration center of gravity. Space shuttle main engine (SSME) TVC is then adequate for control during this period of time. The same requirements described persist at nominal staging. In the second option, the ASRM's on the interstage, no TVC or reorientation of the ASRM's thrust vector is required at any time. Immediately (t = 3 sec) after liftoff, the SSME TVC becomes effective and provides flight control. The control mode is similar for this option at max $\bar{q}$ and at nominal staging as well.

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◢ Abort Performance Comparison

3-472K Lb Vac Thrust MPS, 109% EPL, 7K Lb Vac Thrust OMS

Series Burn PFB Abort SRM's on Interstage

Lift-off Staging Insertion
100 Sec 200 300 400
| RTLS (O+T Flt) | DR | OA ← Orbit |

Series Burn PFB Abort SRM's on Orbiter

Lift-off Staging Insertion
100 Sec 200 300 400
| RTLS (O+T Flt) | DR | OA ← Orbit |
RTLS (Orb Alone Flt)
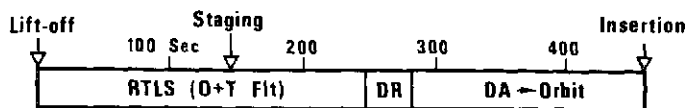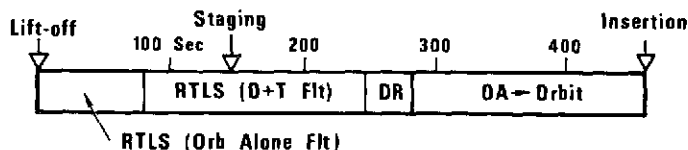
Parallel Burn PFB or SRMB Abort SRM's on Orbiter

Lift-off Staging Insertion
100 Sec 200 300 400
| RTLS (O+T Flt) | DR | OA ← Orbit |
RTLS (Orb Alone Flt)
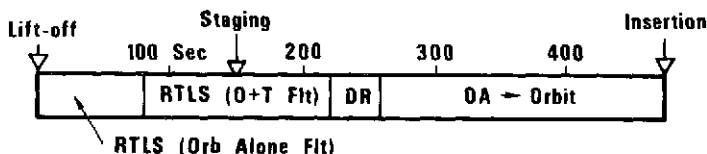
22SV17962
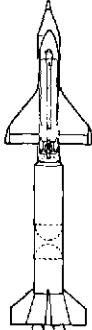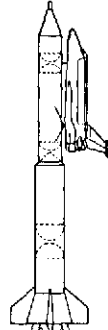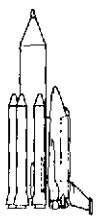
All the configurations have essentially the same abort performance
in that all have the capability to return to the launch site, up to staging.
At staging, with consideration of one engine out, all configurations
also have the capability to return to the launch site up to approximately
250 seconds for the series-burn systems and to approximately 210
seconds for any parallel-burn systems. At this point in the sequence,
there is a gap wherein the vehicle cannot return to the launch site nor
can it be injected into a once-around return orbit. Thus, a downrange
landing requirement persists. After this gap, all configurations can
be inserted into a trajectory that will take them once around to the
launch site, or into a degraded mission capability orbit. It is noted
that this performance is for a 472,000 vacuum thrust orbiter main
propulsion system with 109 percent emergency power level (EPL)
and an orbital maneuvering subsystem (OMS) thrust level of 7000.

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Elimination of Down Range Landing

(One Engine Out)

Recommendation
EPL 116%
OMS Thrust 9600 Lb

40K Payload Polar
11.5K OMS Propellant

Delta Cost ($M) vs Percent Normal Power Level (EPL)

OMS Thrust (K Lb) vs Percent Normal Power Level (EPL)
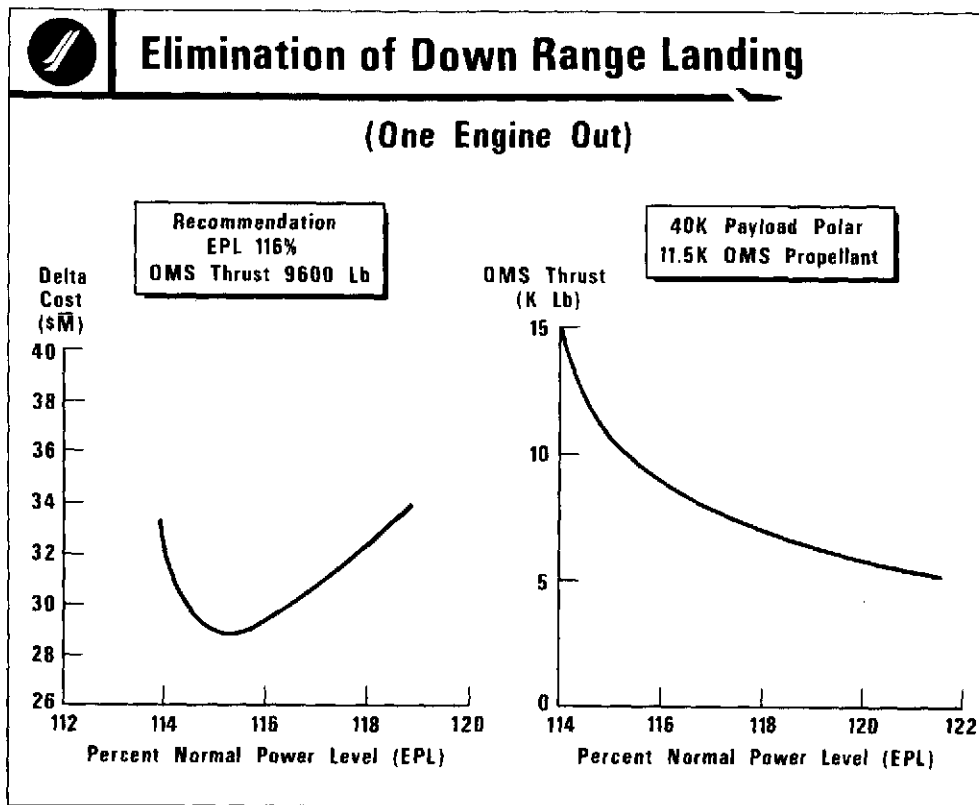
22SV17961

Various options are available to eliminate the requirement for a down-range landing in the case of a one-engine-out abort. The chart illus- trates a tradeoff of orbital maneuvering subsystem (OMS) thrust level versus main engine emergency power level (EPL) where it is shown that a cost-effective solution through variations of these two parameters would result in an OMS thrust level of 9600 pounds and an EPL of 116 percent. Another option not illustrated is the use of the nominal main propulsion system but with excess propellant loaded in the external oxygen-hydrogen tank (EOHT) to provide a contingency specifically for abort sequences. Through the use of this additional propellant, the requirement for higher than nominal EPL (109 percent) may be eliminated. The amount of propellant required to close the down- range landing gap will be less than 2000 pounds for a main propulsion system with 472,000 pounds thrust engines. This amount of propel- lant will have a minimal effect on the overall vehicle size.

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Abort Capability Comparison

	A SRM ON INTERSTAGE	ASRM ON ORBITER	ASRM ON ORBITER
	PFB SERIES SYS	PFB SERIES SYS	SRM PARALLEL SYS 120 IN. OR 156 IN.
WEIGHT - LB	54,000	94,000	94,000
ORBITER SCAR WT - LB	0	700	700
TVC FOR PAD CLEARANCE		±3°	±3°
SEPARATION ORBITER — PAD	ORB & TANK	ORBITER ALONE	ORBITER ALONE
SEPARATION ORBITER — MAX q	ORBITER & TANK	SAME AS PAD OR ORBITER & TANK	SAME AS PAD OR SRM JETTISON
SEPARATION ORBITER — STAGING	ORBITER & TANK	ORBITER & TANK	SRM JETTISON
BOOSTER	SINGLE PLANE	SINGLE PLANE	SRM LINKAGE & THRUST TERMINATOR
ABORT SYS COST $ M̄	298	293	293

22SV18044B

A comparison between the three options discussed is presented. It is seen that the weight of the abort solid rocket motor (ASRM) system on the series burn interstage is on the order of 54,000 pounds and is the lightest system. It is also noted, however, that the weight of the system is inconsequential inasmuch as the abort system pays for itself in performance in all cases. For the systems where the ASRM is mounted on the orbiter, an orbiter penalty will be incurred and will be on the order of 700 pounds. Also, TVC and two-positions for the thrust vector are required for the ASRM.

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Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
• SRM TVC Requirements	• <u>Required</u> for All Systems
• Stage Separation	• <u>Series Burn</u> - State-Of-The-Art <u>Parallel Burn</u> - Intensive Effort Required
• SRM Thrust Termination Requirements	• <u>Required</u> for Abort on All Systems
ABORT	
• Mode	• <u>Series Burn</u> - ASRM on EOHT Interstage <u>Parallel Burn</u> - ASRM on Orbiter
• Control	• <u>Series Burn</u> - MPS TVC <u>Parallel Burn</u> - ASRM TVC With Two Positions Plus Orbiter Aerosurfaces
• MPS Requirements	• 116% EPL OMS Thrust, 9.5K Lb
ENVIRONMENT IMPACT	
TEST & FACILITY IMPACT	

22SV1B246B

The selected abort modes, control requirements, and main propulsion subsystem requirements are summarized as follows: for the series-burn system the abort solid rocket motor (ASRM) on the external oxygen-hydrogen tank (EOHT) interstage was selected for the baseline. For the parallel-burn systems, the ASRM system must be on the orbiter. In the series-burn systems, the main propulsion system provides all of the control during the abort sequence. In the parallel burn system, the ASRM thrust vector control plus aerosurfaces on the orbiter provide control. To eliminate the downrange landing, either an increased EPL on the main propulsion system to 116 percent combined with an OMS thrust level of 9500 pounds must be utilized or additional propellant up to approximately 2000 pounds must be included in the orbiter EOHT.

ENVIRONMENT

22SV18214

Because of the significant differences between the parallel-burn
SRM systems and well-known series-burn liquid systems, it was
desired to investigate the impact of the induced environment in
terms of acoustics, heating, and exhaust products on both the launch
vehicle and the surrounding ecology.

Orbiter Base Heating PFB Series Burn

Heating Rate, q̇
(BTU/Ft² - Sec)

$T_W = 460°\ R$

Pre-MPS Ignition Base Convection
(Orbiter & External Tank Base Surfaces)

Orbiter MPS Operation
Base Convection

0.5
0.4
0.3
0.2
0.1
0

0   100   200   300   400   500   600
Time (Sec)

22SV17921
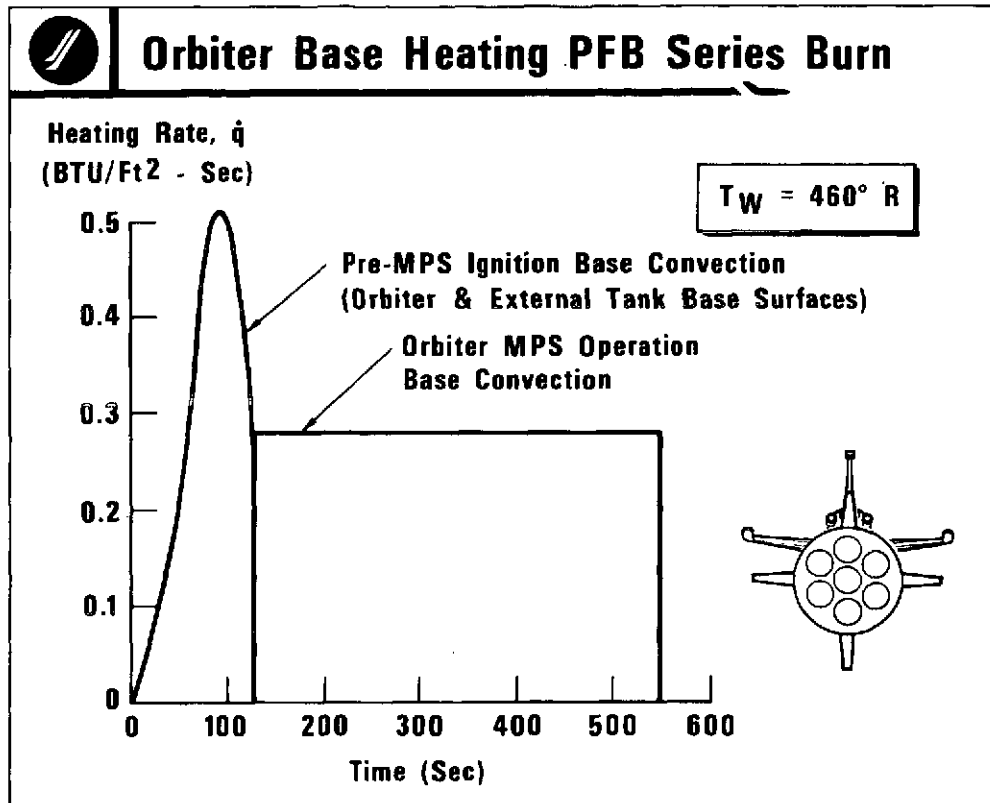
The firing of the pressure-fed booster engines has no significant effect on the orbiter. The chart illustrates base convection due to aerodynamic recirculation and then the base convection due to orbiter engine firing. These are analytically derived results.

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Orbiter Base Heating

Heating Rate, $\dot{q}$
(BTU/Ft^2 - Sec)

120 in. SRM Parallel Burn

——— MPS Base Convection

– – – SRM Exhaust Plume Solid Particle Radiation

Time (Sec)
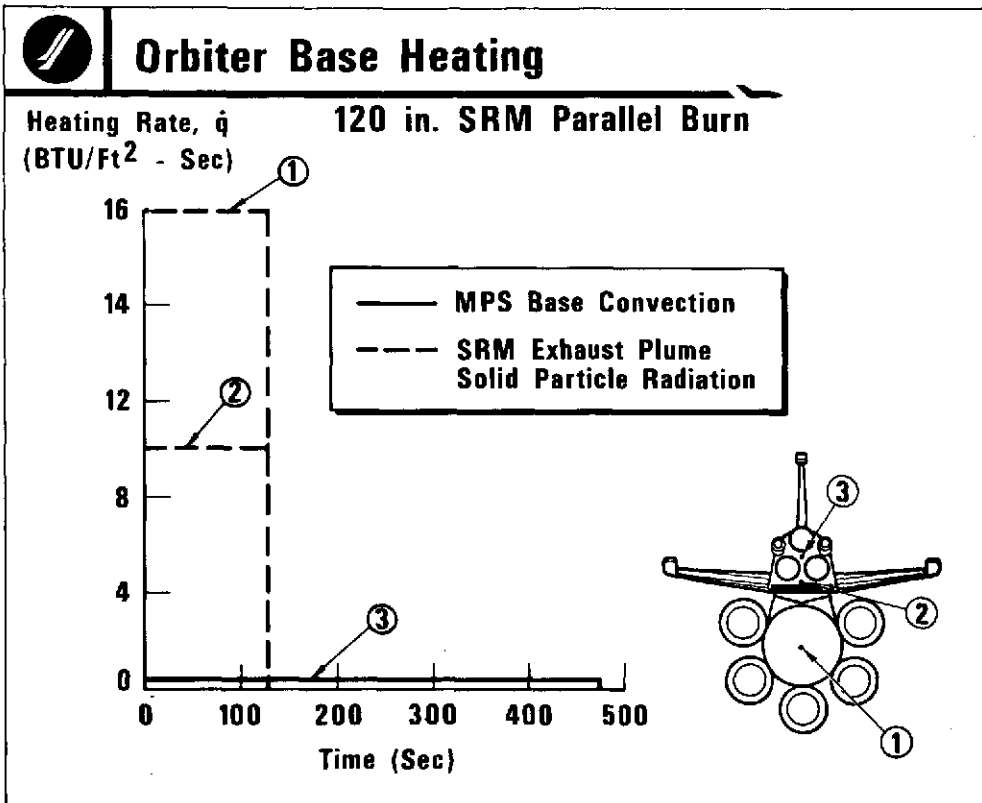
Parallel-burn systems with solid rocket motors create a high base heating environment in both convection for recirculation and radiation for solid particles in the exhaust plume. The accompanying chart shows that the heating rate on the base of the external oxygen-hydrogen tank (EOHT) is significantly higher than during the pressure-fed booster (PFB) series-burn operation where the tank base is shielded by the interstage. Likewise, the heating rate in the base of the EOHT below the lower two nozzles is significantly higher than in the PFB configuration. No significant impact is anticipated in the area between the three orbiter nozzles.

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156 in. SRM Parallel Burn

Heating Rate, $\dot{q}$
$(BTU/Ft^2 - Sec)$

MPS Base Convection

**--- SRM Exhaust Plume
Solid Particle Radiation**

Time (Sec)
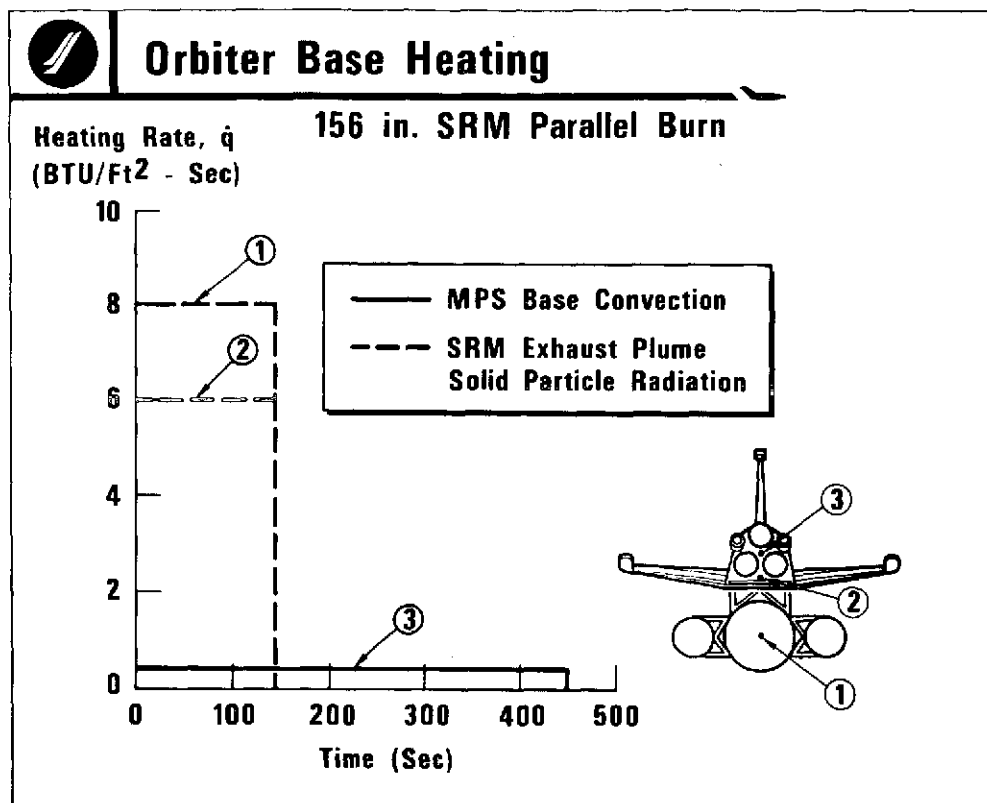
22SV17922A

The anticipated base environment in the 156-inch solid rocket motor (SRM) parallel burn configuration is lower than in the 120-inch SRM parallel burn configuration because of the orientation of the two 156-inch nozzles compared to the clustered arrangement of the 120-inch SRM motor nozzles. Again, the base environment at locations 1 and 2, that is, the back of the external oxygen-hydrogen tank (EOHT) and the lower part of the orbiter base, is significantly higher than in the pressure-fed booster series-burn situation.

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Effect of SRM on Orbiter Base Heat Shield

Maximum Ascent Temperatures (F)	①	②	③
PFB	700	620	700
120 in.	1070	1310	700
156 in.	1020	1220	700

• Present Design Not Impacted by Increased Temperatures
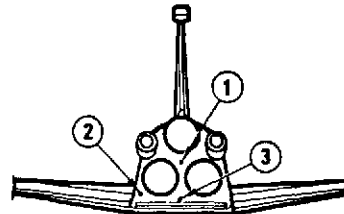
Increased Base Heating From SRM Plume Radiation

Base Heat Shield Δ Weight	
PFB	0
120 in.	40 Lb
156 in.	30 Lb

• *NEGLIGIBLE DESIGN IMPACT*

22SV17982

Heat shield weights for all three systems were computed. It is seen that almost negligible additional heat shield weight is required to compensate for the radiation from the solid rocket motor plumes because the design environment arises during entry, not ascent.

Ascent Aero Heating Impact

PFB Series Burn
Radiation Equilibrium Temperatures

910 F
1050 F
385 F 375 F
2100 F
780 F 650 F 550 F
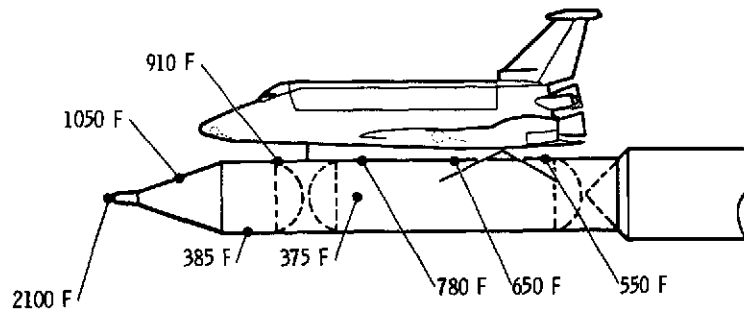
22SV17969A

Radiation equilibrium temperatures were computed on the external oxygen-hydrogen tank for the pressure-fed booster series-burn ascent environment. The results of these calculations are shown in the accompanying chart.

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120 in. SRM Parallel Burn
Radiation Equilibrium Temperatures

780 F

910 F

1050 F

2100 F

385 F

480 F

650 F

550 F
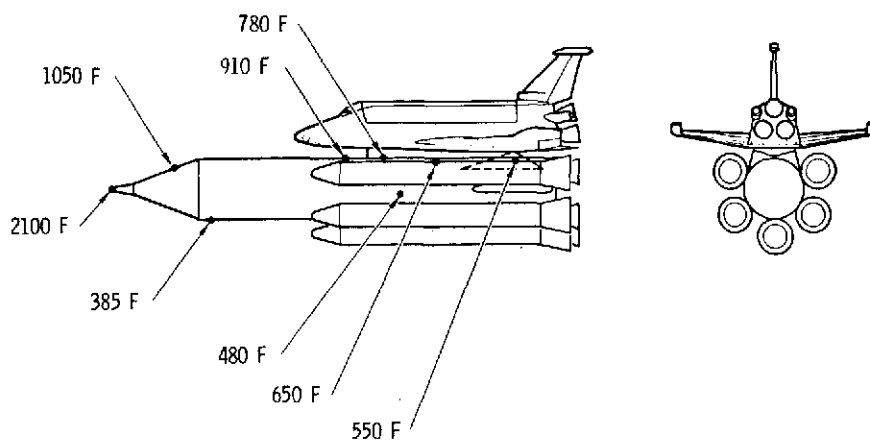
22SV17971C

In a manner similar to that for the pressure-fed booster (PFB) series-burn situation, radiation equilibrium temperatures were computed on the external oxygen-hydrogen tank for a 120-inch solid rocket motor parallel-burn ascent. A comparison indicates that these temperatures are not significantly higher than those encountered during a PFB series-burn ascent, although there are special areas of interference heating because of the attachment of the five rocket motors.

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Ascent Aero Heating Impact

156 in. SRM Parallel Burn
Radiation Equilibrium Temperatures

910 F

560 F

1050 F

2100 F

385 F

375 F

780 F

650 F

550 F
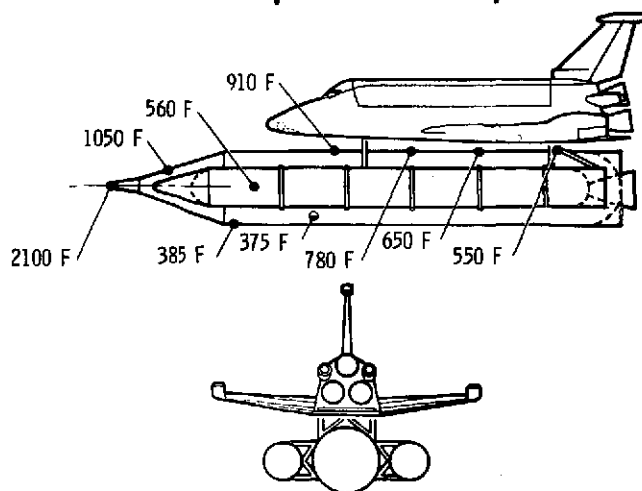
22SV17967C

Radiation equilibrium temperatures were computed on the external oxygen-hydrogen tank (EOHT) during the ascent of the 156-inch solid rocket motor parallel burn configuration. Again, the temperatures encountered were not significantly different than in the PFB series-burn configuration although special areas of interference heating on the EOHT can be expected because of the attachment of the solid motors.

Ascent Aero Heating TPS Reqmts

PFB Series Burn

0.66 Avcoat 5026-39

0.40 Cork

910 F

0.45 Cork

SOFI-Sidewall & Bulkheads

0.39 Cork

0.37 Cork
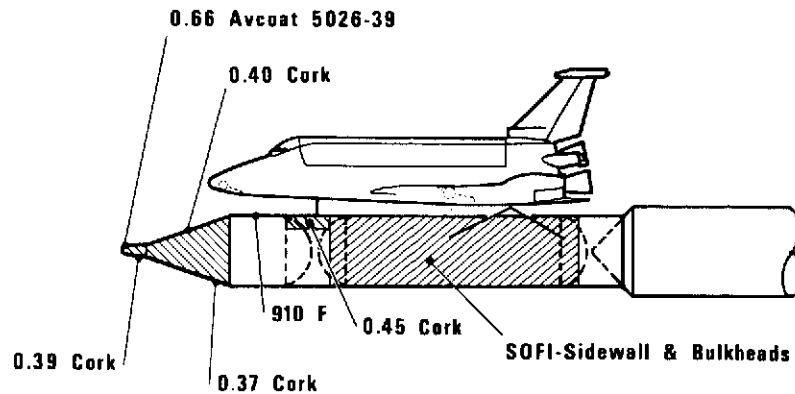
Baseline

22SV17970C

The accompanying chart illustrates the insulation on the pressure-fed booster series-burn external oxygen-hydrogen tank (EOHT). Shown are cork ablator on the nose cone of the EOHT and spray-on foam insulation on the hydrogen tank sidewalls and bulkheads. Cork also is used in special interference areas.

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Ascent Aero Heating TPS Reqmts

120 in. SRM Parallel Burn

910 F
0.45 Cork
0.40 Cork
0.39 Cork

0.66 Avcoat
6026-39

0.32 Cork

SOFI-Sidewall
& Bulkheads

0.42 Cork
Bonded
to Mech
Fastened Face
Sheet Over
SOFI
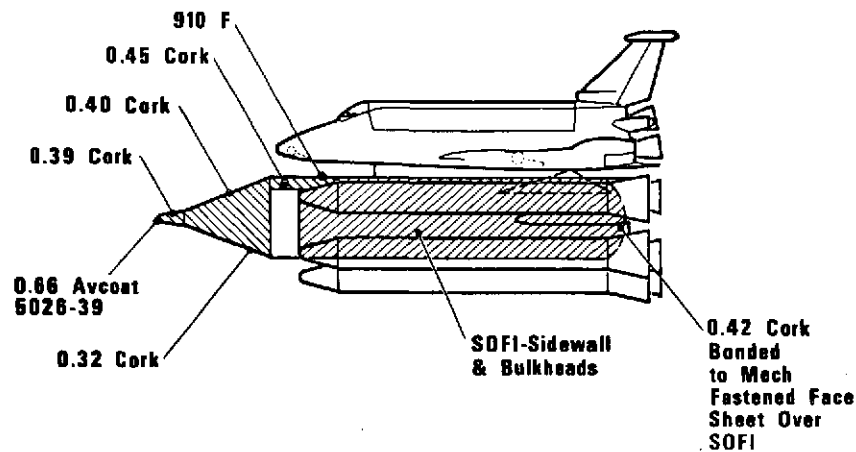
Δ TPS Weight: +2850 Lb

22SV17968D

In this configuration the installation of thermal protection system (TPS) on the external oxygen-hydrogen tank (EOHT) is similar to that in the pressure-fed booster series burn. However, additional insulation is required on the base of the EOHT where cork is bonded to a face sheet that is mechanically fastened over the spray-on foam insulation. Substantial increase in the weight of the TPS is incurred (2850 pounds).

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Ascent Aero Heating TPS Reqmts

156 in. SRM Parallel Burn

0.25 Cork Bonded to
Mech Fastened Face
Sheet Over SOFI

0.40 Cork

0.66 Avcoat
5026-39

910 F

0.39 Cork

0.37 Cork

0.45 Cork

0.29 Cork

SOFI-Sidewall
& Bulkheads

0.39 Cork Bonded
to Mech Fastened
Face Sheet Over
SOFI

Δ TPS Weight = +1450 Lb
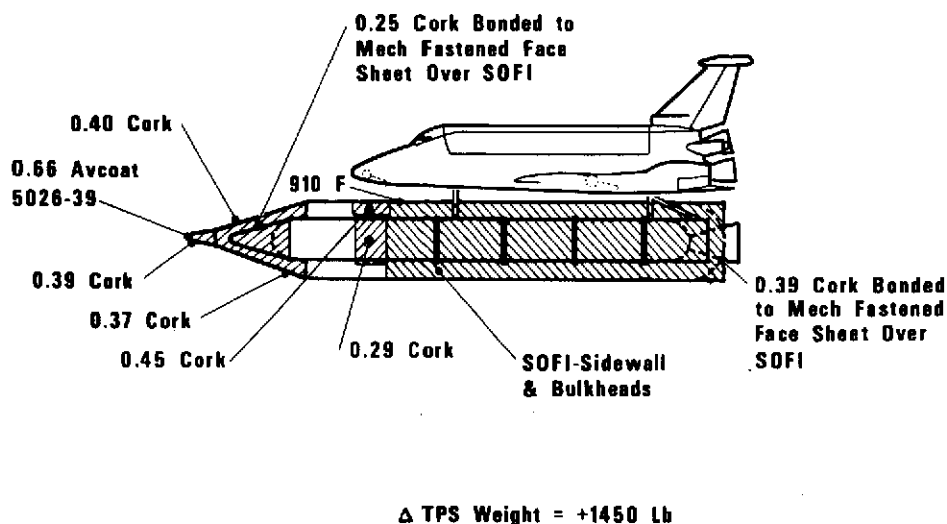
22SV17966D

The insulation requirements for the external oxygen-hydrogen tank (EOHT) of
the 156-inch solid rocket motor parallel burn configuration are illustrated.
Again, cork bonded to a face sheet mechanically fastened over the spray-on
foam insulation (SOFI) is required on the aft bulkhead of the EOHT. An increase
in thermal protection system weight of 1450 pounds is incurred.

159 db   160 db

159 db   160 db

Baseline Wt

22SV17978B
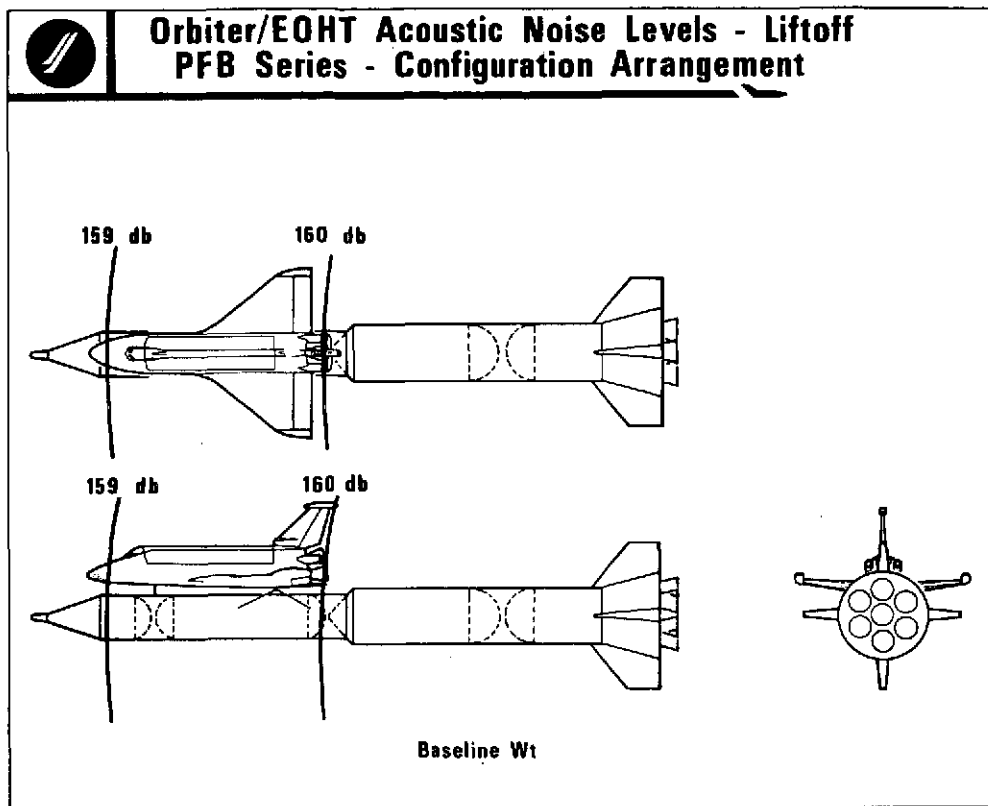
The anticipated noise level from the orbiter base to the orbiter crew
compartment is shown. The levels vary from 160 db to 159 db over the
orbiter vehicle.

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160 db   161 db   164.5 db
166 db

166.5 db

166 db

164.5 db   166.5 db

161 db

160 db

Δ to Baseline: Wt = +965 Lb
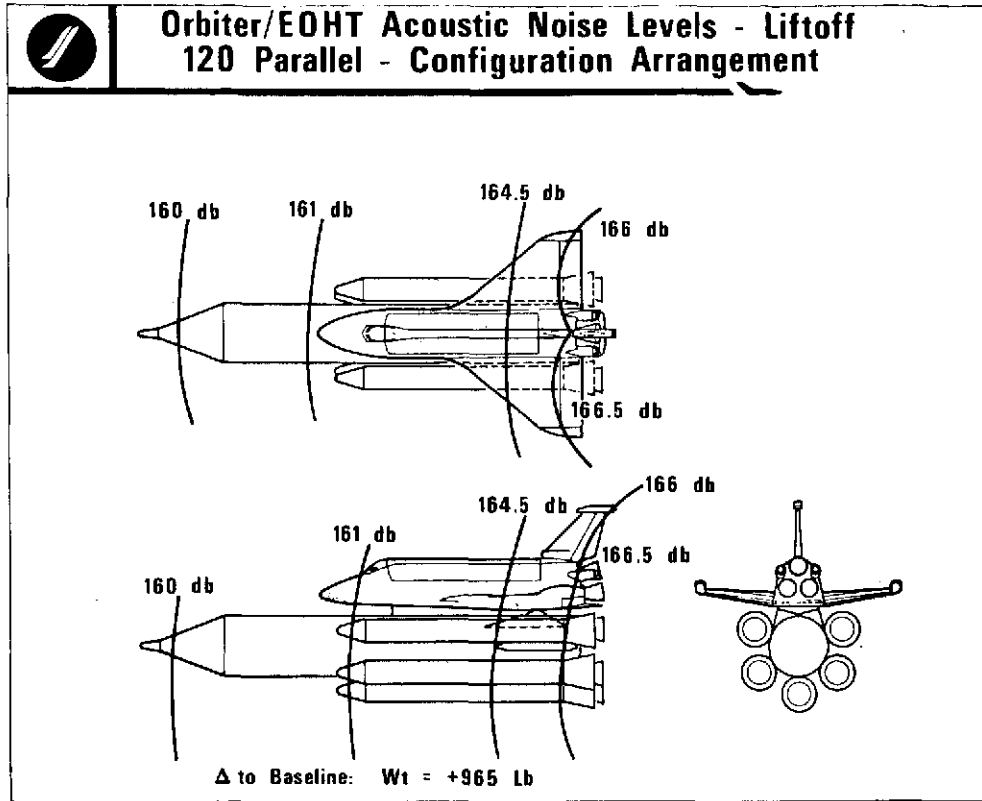
22SV179808

This chart illustrates the noise levels to be encountered from the base of
the orbiter to the tip of the external oxygen-hydrogen tank in a 120-inch
parallel-burn configuration. It is seen that the noise levels at the orbiter
base are significantly higher than in the pressure-fed booster series-
burn configuration and the overall noise level encountered by the orbiter
also is noticeably higher. An increase in orbiter weight of 956 pounds
is required.

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Orbiter/EOHT Acoustic Noise Levels - Liftoff
156 Parallel - Configuration Arrangement

162.5 db

161.5 db

166 db   167.5 db

168 db

167.5 db

162.5 db   166 db

167.5 db

168 db

Δ to Baseline: ΔWt = +1160 Lb

22SV179798
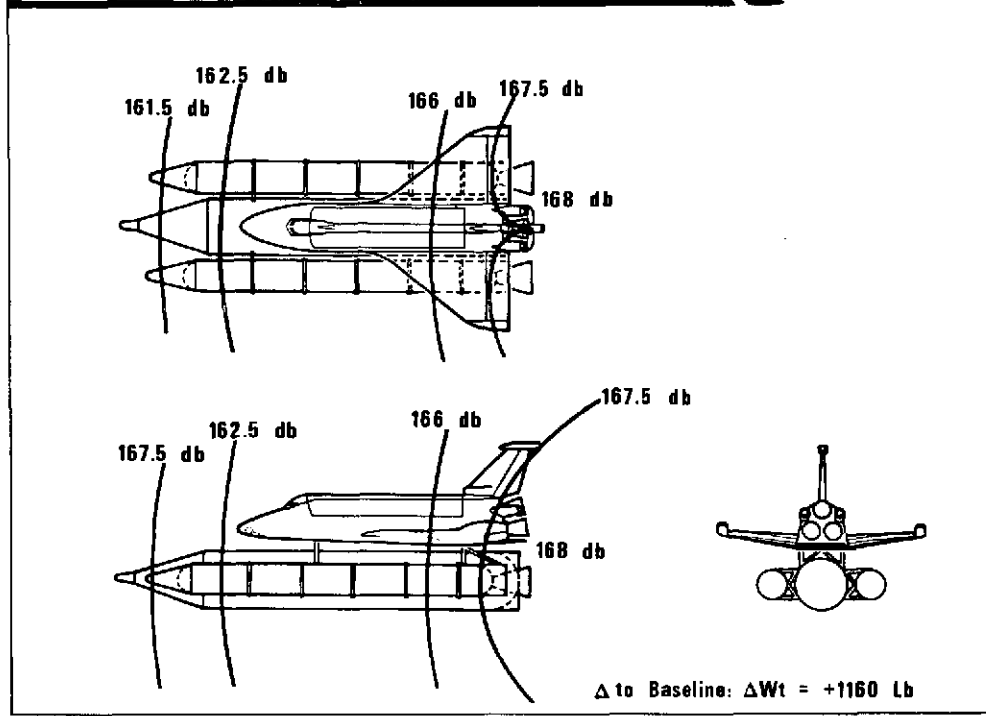
The noise levels for a 156-inch parallel burn configuration were
calculated. Again, the noise levels for this configuration are signifi-
cantly higher than for the pressure-fed booster configuration, and range
from 168 db at the orbiter base to a 161.5 db at the tip of the external
oxygen-hydrogen tank. In both the parallel-burn configurations con-
sidered (i. e., 120-inch and 156-inch solid rocket motors), delta weight
to the orbiter to compensate for the increased noise level was estimated
to be on the order of 1165 pounds.

Orbiter Aerodynamic Noise - Ascent

PFB Series - Configuration Arrangement Transonic & Qmax

155 (157)

dB LEVEL FOR $-8^0 \le \alpha \le +8^0$; $-5^0 \le \beta \le +5^0$

dB LEVEL FOR $\alpha = \beta = 0$

* 164
* 155

154 (155)

160 (163)

152 (160)
153

152

NO STRUCTURAL IMPACT

APOLLO EXTERNAL dB = 160
INTERNAL dB = 110-115
* ORBITER INTERNAL dB = 120-125 EST
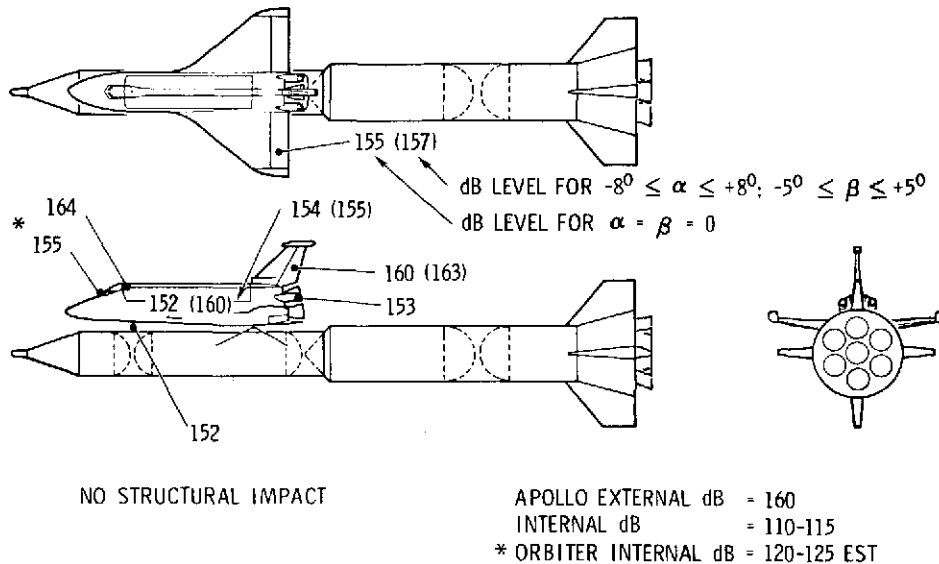
22SV17977

The chart illustrates the aerodynamic noise anticipated during ascent of a pressure-fed booster series-burn configuration. Noise levels for angles of attack from -8 to +8 degrees and yaw angles from -5 to +5 degrees were calculated and are presented, together with the noise levels anticipated for α and β equal to 0 degrees.

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dB LEVEL FOR $-8^0 \le \alpha \le +8^0$; $-5^0 \le \beta \le +5^0$
dB LEVEL FOR $\alpha = \beta = 0$ —

157 (162)

152 (160)
164
155
156
160 (163)
161 (165)

PFB SERIES < 156 PARALLEL < 120 PARALLEL

22SV18048A
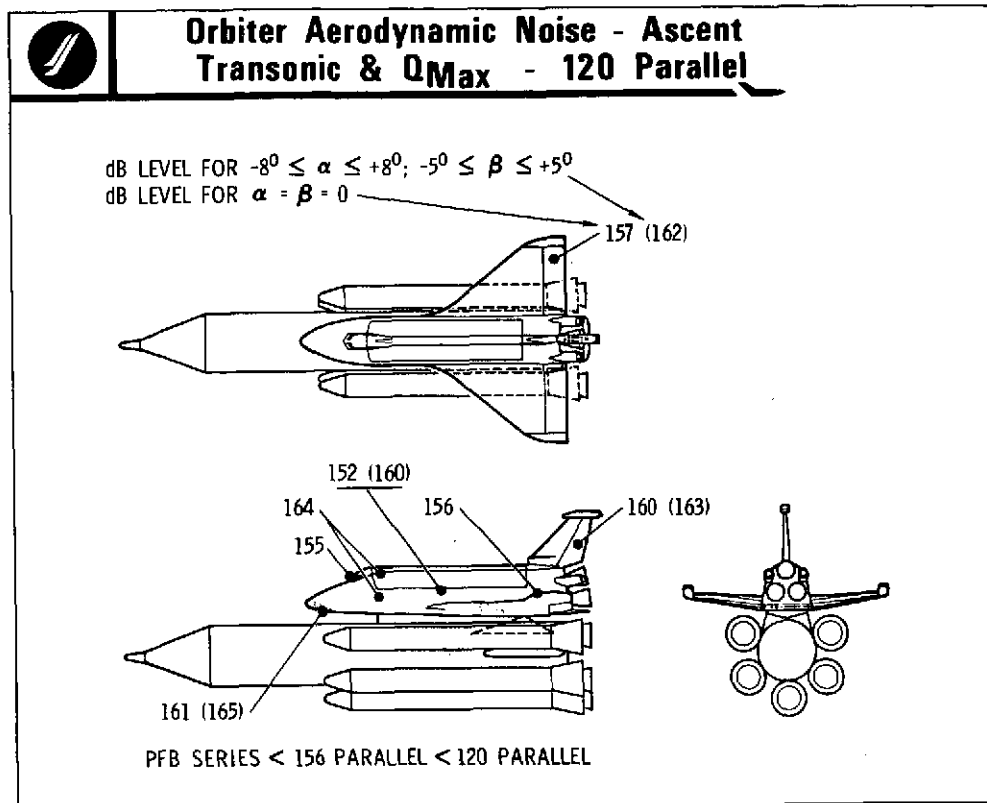
Aerodynamic noise levels for a parallel-burn configuration with 120-inch
solid rocket motors were estimated. In general, those at the forward
end of the orbiter are similar in intensity to those encountered during
a pressure-fed booster series-burn configuration during ascent.
However, at the aft end of the orbiter, the noise levels anticipated on
the orbiter elevons are higher in the parallel-burn configuration. It is
anticipated that a similar situation occurs with a 156-inch solid rocket
motor configuration. No significant impact is anticipated because of
aerodynamic noise.

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Sound Pressure Level at KSC

(All Configurations)

Exhaust
Directions

99 db
102 db
105 db
108 db
111 db
114 db
120 db
117 db

Titusville 106 db

Pad LC-39

Exhaust
Directions

Vab 120db

Cocoa
100db

Cocoa
Beach

Levels are ≈1 db Below Saturn V

REF. MSFC. S&E AERO-AU (7/8/71)

225V1796BA
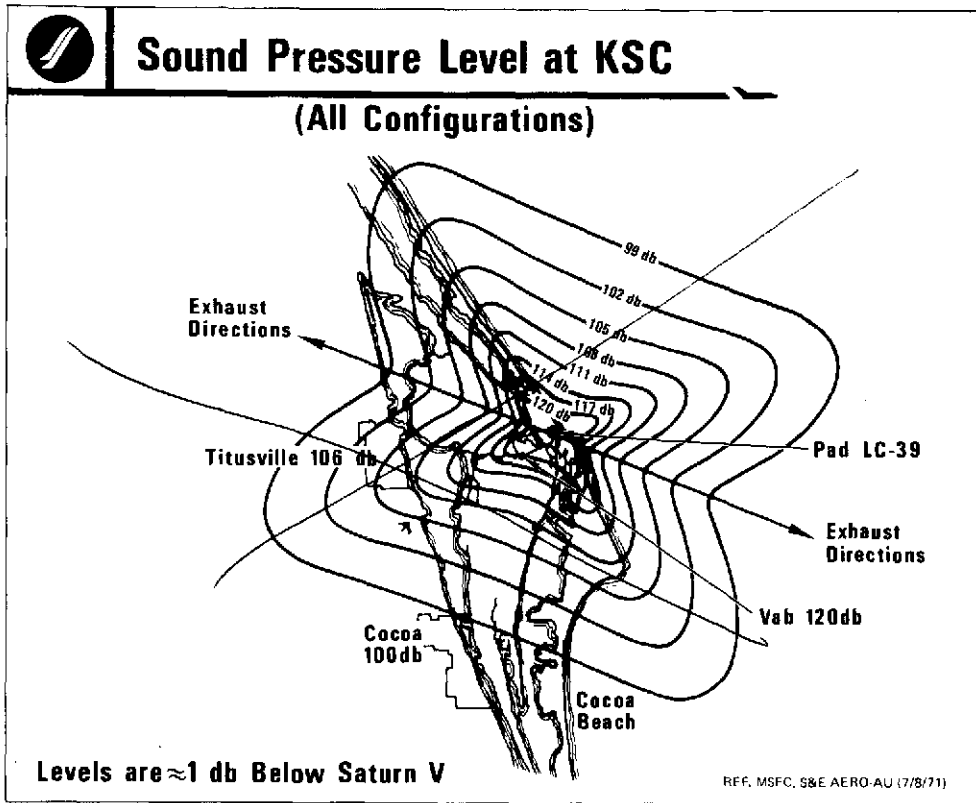
The accompanying chart shows a contour map of ground-imposed db levels during the launch of any of the configurations considered. It is anticipated that the levels will be 1 db below those imposed by the launch of the Saturn V at pad LC-39.

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Air Pollution

Preliminary Estimates of Possible Dosages at Titusville Under Unfavorable Meteorological Conditions*

	Normal Launch		Abort	
	Solid	O$_2$/Propane	Solid	O$_2$/Propane
Carbon Monoxide				
Federal Standard Continual Exposure (PPM)		50		
Total Dosage For Short Dur (PPM-Min)		At Least 2,100		
● Estimated Peak Concentration (PPM)	48	62	52	65
● Estimated Total Dosage (PPM-Min)	210	270	230	280
Hydrogen Chloride				
Federal Standard Continual Exposure (PPM)		5 Maximum		
Total Dosage for Short Duration (PPM-Min)		300 (Tentative)		
● Estimated Peak Concentration (PPM)	31	0	34	2
● Estimated Total Dosage (PPM-Min)	140	0	150	10
Particulates				
Federal Standard Exposure (MG/M^3)		0.26 for 24 Hr, 0.075 Continual		
Total Dosage for Short Duration (MG-MIN/M^3)		375 (Tentative)		
● Estimated Peak Concentration (MG/M)	65	0	70	
● Estimated Total Dosage (MG-Min/M^3)	280	0	310	20

* During 18 Titan Launches The Cloud Has Never Reached The Ground

22SV18045C

A table of exhaust products compared to federal standards for continual exposure is presented. It is seen that in no case does the estimated total dosage exceed the federal standard for total dosage for short duration. It is noted on the chart that during 18 Titan launches, the exhaust plume cloud has never reached the ground but has dissipated in the air. It is also seen that the oxygen propane exhaust from a pressure-fed booster is noticeably cleaner than the exhaust of solid propellant motors particularly in terms of hydrogen chloride and particulates.

Pollutant Production Characteristics of Alternative Launch Vehicles

Mode	Booster Type	Burn Sequence	Solid Rocket Size	Maximum Total Discharge of Pollutants Before Reaching 2000 Ft Altitude (1000 Lb)		
				Carbon Monoxide	Hydrogen Chloride	Particulates
Normal Launch	Solid Rocket Motors	Series	120	161	138	187
			156	159	136	185
		Parallel	120	100	86	116
			156	83	71	97
	O2/Propane Pressure-Fed	Series		206	0	0
		Parallel		186	0	0
Abort	Solid Rocket Motors	Series	120	174	149	202
			156	172	147	200
		Parallel	120	110	94	127
			156	93	80	108
	O2/Propane Pressure-Fed	Series		218	11	14
		Parallel		198	11	14

22SV18131B

The chart lists the characteristics of the exhaust products in terms of total pounds discharged before reaching a 2000-foot altitude for the solid rocket motors considered and the oxygen propane exhaust of the pressure-fed system during a normal launch and an abort.

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Impact Areas

PFB, Series Burn-Normal Missions

RECOVERABLE PFB

ASRM CASES PLUS INTERSTAGE

125 N MI

170 N MI

225V18049
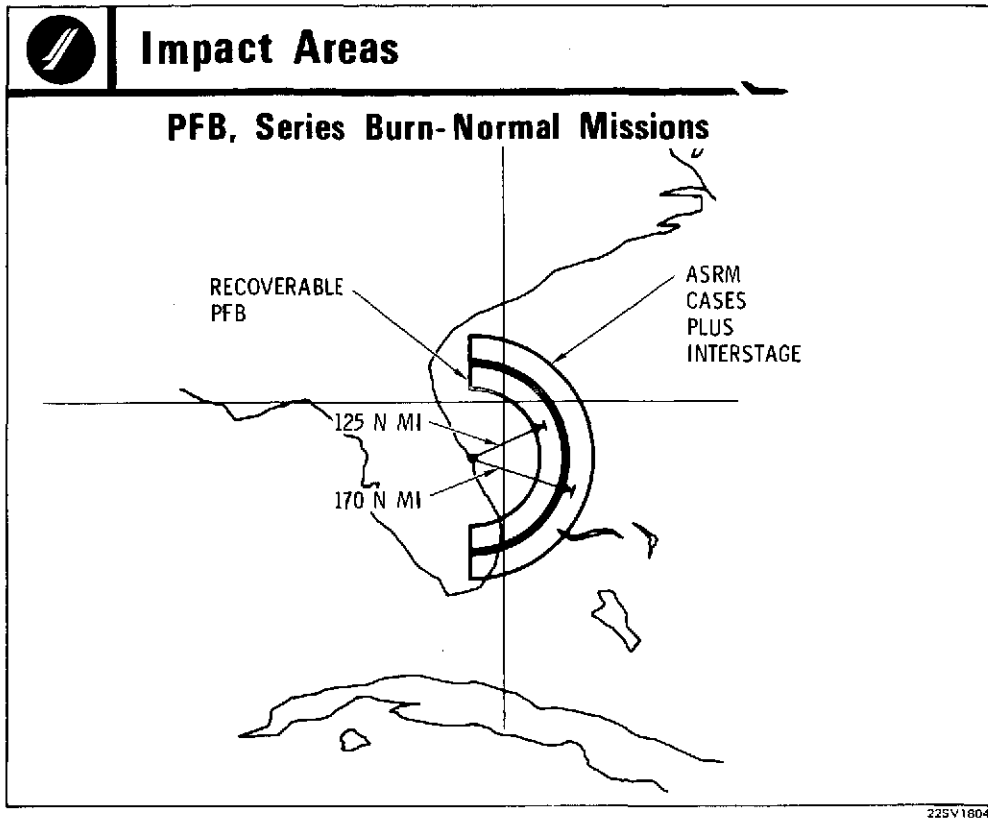
Illustrated on the chart are the fallout areas for a pressure-fed booster (PFB) series-burn normal mission of the ASRM motor cases and inter-stage as well as the recoverable PFB. It is seen generally that, except for southward launches, no problem should be encountered in the impact of these devices.

Impact Areas

120 in. SRMB, Parallel Burn-Normal Missions

BOOSTER
120 IN.
SRM CASES

ASRM
CASES

160 N MI

240 N MI
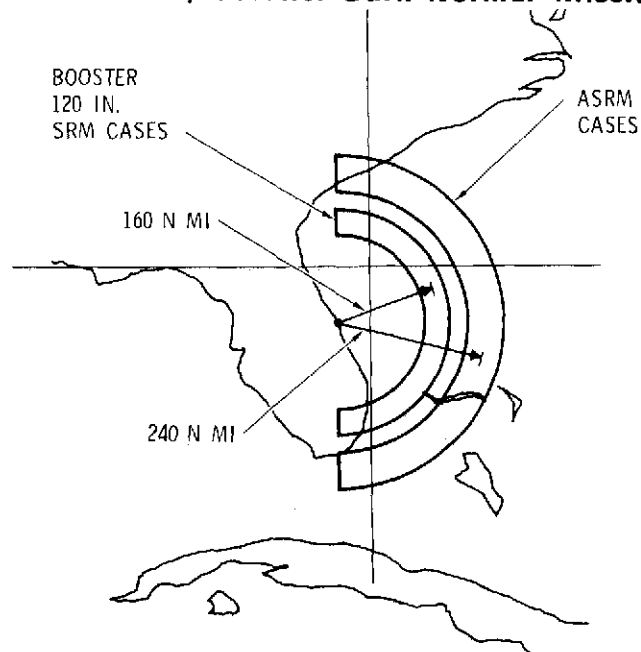
22SV18050

Shown are the impact areas for the 120-inch solid rocket motor (SRM) cases and the abort solid rocket motor (ASRM) cases for a 120-inch SRM parallel burn vehicle. Here, it is seen that the Grand Bahama island falls in the impact range as does the southernmost tip of Florida for a southern launch and the coast of Georgia for a direct northern launch. Except for these three instances, no problems should arise.

Impact Areas

156 in. SRMB, Parallel Burn-Normal Missions

ASRM
CASES

BOOSTER
156 IN. SRM
CASES

185 N MI

280 N MI

22SV16051
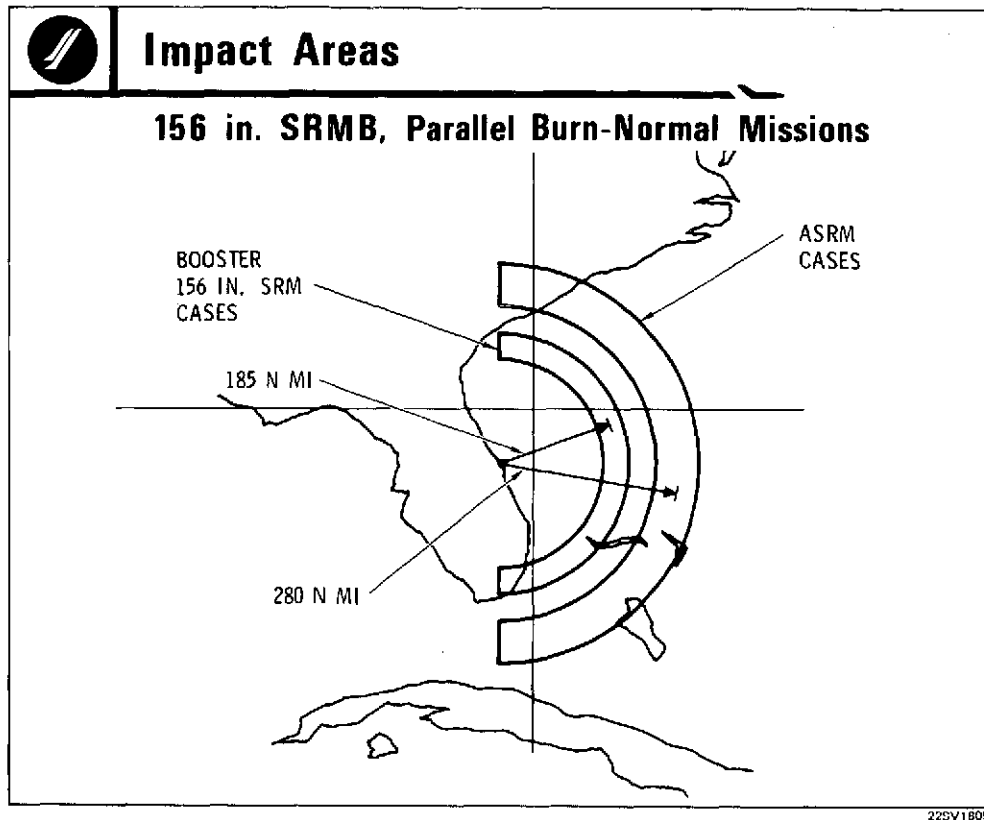
Shown are the impact areas for the abort solid rocket motor cases and
the 156-inch solid rocket motor cases for a 156-inch parallel-burn
vehicle. Fallout area pattern is similar to that for the 120-inch parallel
burn system except that the abort solid rocket motors impact further
downrange.

h / K FT

100

BOUNDARY FOR
1.5 PSF SURFACE
OVERPRESSURE

SHAPED ENTRY TRAJECTORY

50

• TRAJECTORY SHAPING TO AVOID SONIC BOOM

0

1   2   3   4

M

64 N MI

ΔP >1.5 PSF
(FOR NOMINAL TRAJECTORY)

• ALTERNATE, TARGET FOR M = 3.0 TO AVOID SONIC BOOM ON LAND

TO AVOID SONIC BOOM

• TARGET FOR 100,000 FT OVER LANDING SITE

OR • RESHAPE ENTRY TRAJECTORY

22SV18130A
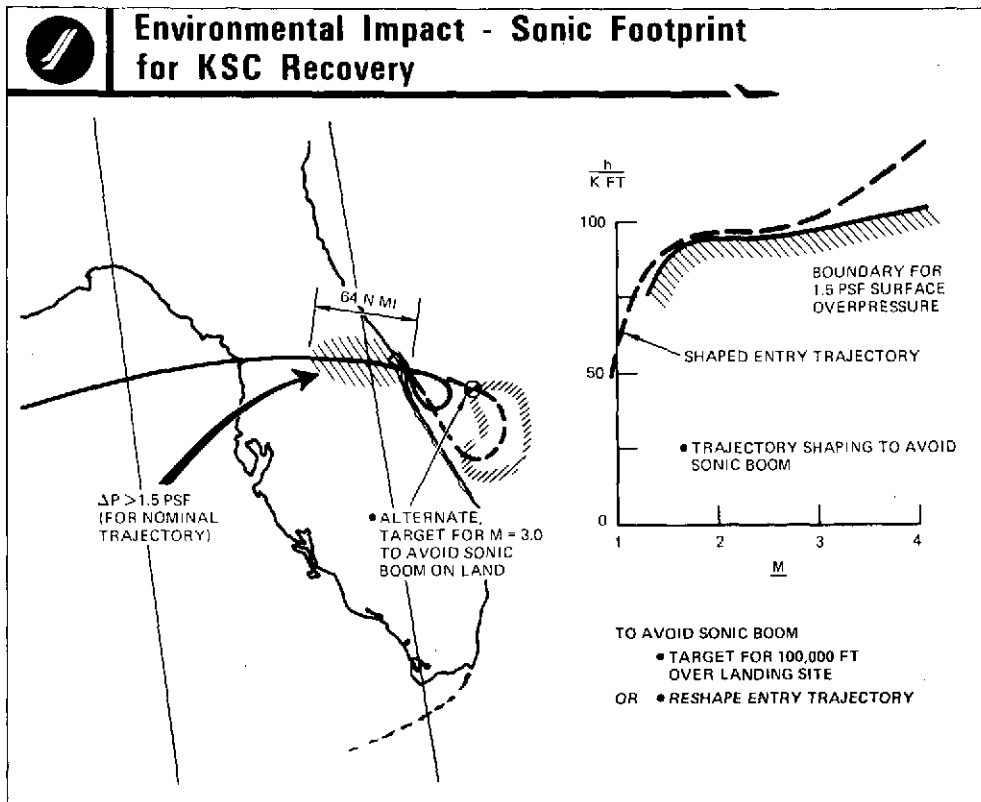
The chart illustrates the sonic footprint imposed by 1.5 psf overpressure during a nominal trajectory. It has been determined that trajectory shaping will be required to avoid imposing this footprint on the continental areas of Florida. An overflight out to sea with return to the landing site may be required to avoid sonic boom on land.

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● | Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
● SRM TVC Requirements	● Required for All Systems
● Stage Separation	● Series Burn - State-Of-The-Art Parallel Burn - Intensive Effort Required
● SRM Thrust Termination Requirements	● Required for Abort on All Systems
ABORT	
● Mode	● Series Burn - ASRM on EOHT Interstage Parallel Burn - ASRM on Orbiter
● Control	● Series Burn - MPS TVC Parallel Burn - ASRM TVC With Two Positions Plus Orbiter Aerosurfaces
● MPS Requirements	● 116% EPL OMS Thrust, 9.5K Lb
ENVIRONMENT IMPACT	● For The 3 Configurations, Environ Acceptable & Not a Discriminator
TEST & FACILITY IMPACT	

22SV18245C

The technical issues discussed so far are summarized on the
accompanying chart. It is noted that the environments imposed by
the pressure-fed-booster and solid-rocket-motor systems are
acceptable on the ground and in impact on the orbiter. Therefore,
this particular issue is not a system discriminator.

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TEST & FACILITIES

22SV18261

A comparison has been made of the test requirements and facility requirements for the pressure-fed-booster series-burn systems and the solid-rocket-motor parallel-burn systems to determine if these requirements become a system discriminator.

Test Program Reqmts Comparison

(Highlights)

Config / Item	PFB Series Burn	156 In. SRM (Parallel Burn)	120 In. SRM (Parallel Burn)	Comments
Booster				
• Structure	Static & GVT	Case Qual	Case Qual	PFB - LITVC
Main Prop. Sys	Eng Dev+Cluster	Segment/Nozzle Dev + Static Firing	Same as 156 in.	
• RCS & Avionics	Recovery Reqmt	None	None	
Retrieval/Recovery	Extensive Testing	None	None	Impact, Chutes, Tow & Refurb
Orbiter-EOHT	Tank Static	More Complex Static	More Complex Static	SRM Mult Attachments
Integrated System				
• Separation B/0	Components- Pyro	1 SRM Test Art	1 SRM Test Art	Mech Reliability
• Dynamics - Modal	Analysis	Full Scale Mated	Full Scale Mated	
Acoustics	Analysis	Cluster Firing	Cluster Firing	
Flight Test	1st Flt Unmanned	Same	Same	
Total Test Complexity	100%	59%	62%	

Test requirements were compared for the three configurations under consideration by first establishing the key items to be tested and then establishing a complexity factor for each. The complexity factor was used to determine the relative difficulty in accomplishing each test program. The most significant differences were, first, to demonstrate recovery capability in the pressure-fed-booster (PFB) systems. No such requirement existed in the solid-rocket-motor (SRM) systems. Also, a main propulsion system test article is required for the PFB, whereas only single-nozzle development testing is required for the SRM's. Finally, full-scale, mated, dynamic testing for modal characteristics is required for the parallel-burn SRM systems because of their complexity, whereas the PFB series-burn system can be determined by analysis. In summary, the solid motor systems are significantly less complex than the PFB series-burn system because of booster system simplicity and elimination of the requirement to demonstrate recovery and retrieval.

Booster Cost Comparison Summary

Facilities
- Test
- Operations

Manpower
- Test
- Activation
- Operations

Support Equip.
- Test
- Operations

Logis & Maint
- Spares
- Transportation
- Fluids & Gases

	PFB	120 in.	156 in.
Total End Items/Support EQT	484	193	155
Total Man Months	64,060	35.233	28.827
DDT&E MM	27,096	14.903	12.195
Δ Facilities Cost Total	0 (Base)	-429.54$\overline{M}$	-549.24$\overline{M}$

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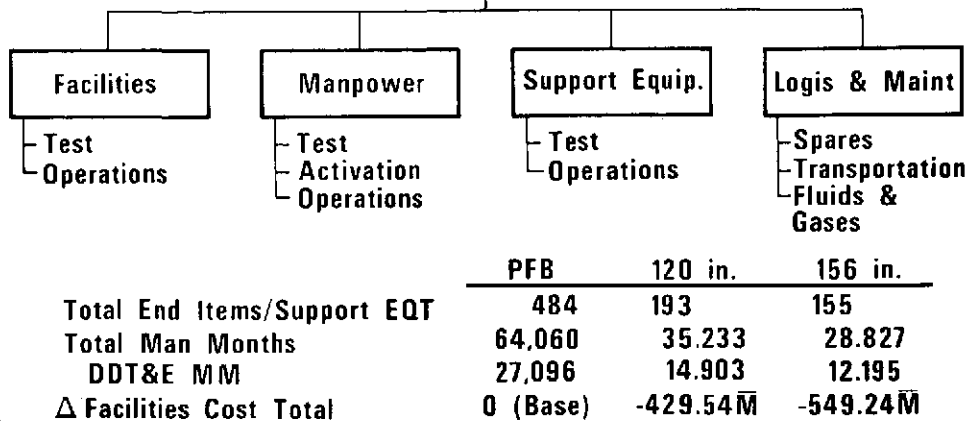
The relative cost was determined for facilities to support each of
the three programs. Also, the man-months required to support
each program were estimated and, finally, the support equipment
requirements were evaluated for each program. Significantly less
support equipment was required for the solid motor systems and
significantly fewer man-months were required to support design,
development, test, and evaluation as well as the total program.

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Technical Issues

ISSUE	FINDINGS
BOOSTER ISSUES	
• SRM TVC Requirements	• Required for All Systems
• Stage Separation	• Series Burn - State-Of-The-Art Parallel Burn - Intensive Effort Required
• SRM Thrust Termination Requirements	• Required for Abort on All Systems
ABORT	
• Mode	• Series Burn - ASRM on EOHT Interstage Parallel Burn - ASRM on Orbiter
• Control	• Series Burn - MPS TVC Parallel Burn - ASRM TVC With Two Positions Plus Orbiter Aerosurfaces
• MPS Requirements	• 116% EPL OMS Thrust, 9.5K Lb
ENVIRONMENT IMPACT	• For The. 3 Configurations, Environ Acceptable & Not a Discriminator
TEST & FACILITY IMPACT	• Test for SRM'S Reduced • Fac Cost for SRM'S Reduced

22SV18247C

In summary, an evaluation of the three systems under consideration revealed that the test facilities, manpower, and support equipment costs for the solid-rocket-motor systems would be significantly less than for the pressure-fed booster systems.

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ORBITER

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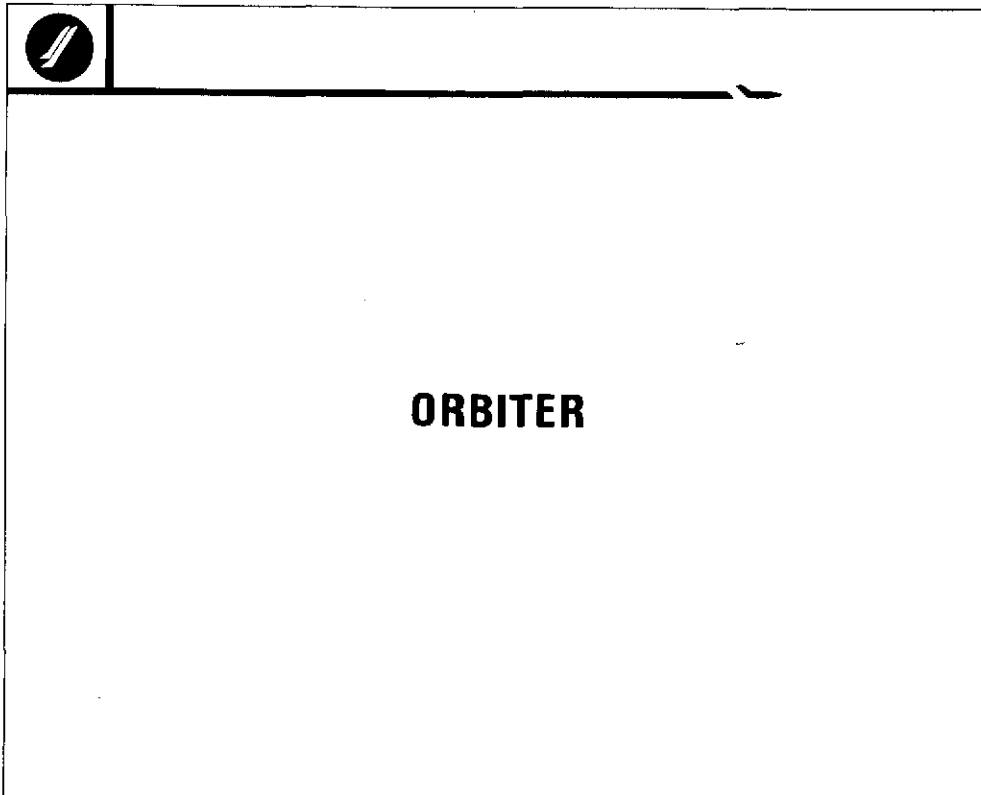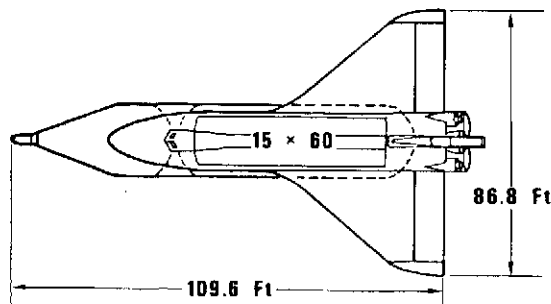
OLOW	1,345	(K Lb)
Tank Dry Wt	48.7	(K Lb)
Landing Wt	194.3	(K Lb)
Theo Wing Area	3,440	(Ft^2)

15 × 60

86.8 Ft

109.6 Ft

32 Ft

133.3 Ft

23.8 Ft Dia
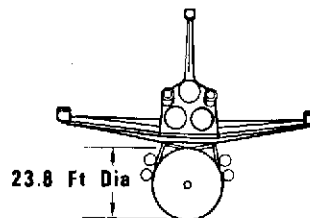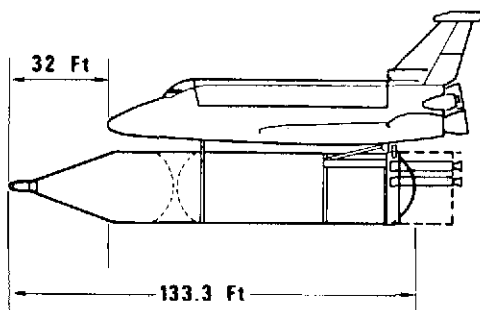
225V17896C

The baseline orbiter and external oxygen-hydrogen tank (EOHT)
arrangement is shown. Total liftoff weight predicted for the orbiter
is 1.345 million pounds. The EOHT is a skin-stringer construction
with LOX forward and LH$_2$ aft. The illustration shows the interstage
between the EOHT and the booster. Mounted on the interstage are
the four abort rockets. The landing weight shown includes a 40,000-
pound payload. The wing is designed for a landing speed of 156 knots.

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Manipulator Station

Flight Deck
(2 Crew)

Engine Compt
Avionics Bay

Engine Compt Access

(4) APU's

RCS Pod

Star Tracker

Payload Bay

Door/Radiator

Manipulator

ABES
Bay

Rudder &
Speed Brake

Airlock

OMS Pod

Cyro Tanks

Main Engine (3)

Fuel Cells

Tank Propellant
System Disconnects

Tank Support Attach

Tank Support Attach

Nose Gear

ABES Propellant

ECLSS

Main Gear

Avionics Bay &
Lower Crew Compt

Side Hatch

Service Panel (Fuel Cells)
in Ldg Gr Well

22SV17895C

The inboard profile drawing illustrates the interior arrangement of the major subsystems. Highlights of the propulsion system shown are all propellant tank disconnects mounted aft. The orbit-maneuvering-system pods are on opposite sides of the aft fuselage, wing-tip- and vertical-tail-located reaction-control-system pods, and the three-engine main-propulsion system. Shown in phantom in the payload bay is the air-breathing engine system kit. The payload bay features manipulators located in a fairing and radiators mounted on the payload bay doors. The crew and passenger station has a forward-mounted air lock and lower avionics bay and crew compartment. A side hatch provides for rapid egress.

⚡ Aft Compt Config — 3 Eng EOHT Orbiter

LH₂ Fill & Drain Disconnect

LH₂ Vent Disconnect

LH₂ Engine Inlet Line

LH₂ Prevalve

LH₂ Feed Manifold

Main Engine Feed Duct

LO₂ Feed Manifold

Main Engines (3) 472 K Lb Vacuum Thrust

LO₂ Feed Orbiter/Ext Tank Disconnect

LO₂ Vent Valves

LH₂ Vent Valves

LH₂ Feed Orbiter/Ext Tank Disconnect

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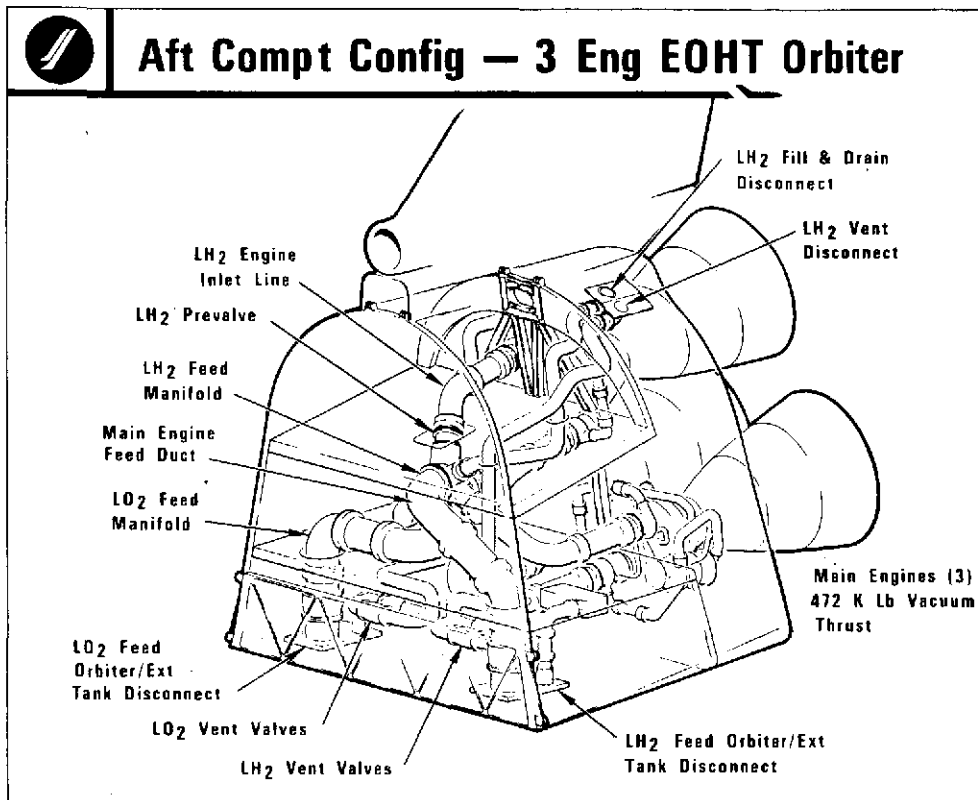
The illustration describes the installation of the three main-propulsion-system engines. Each has 472,000 pounds vacuum thrust. The illustration features the main propellant feed systems, the feed system disconnects to the external oxygen-hydrogen tank (EOHT), the main prevalves, and the LH₂ fill and drain disconnects and vent disconnects. The illustration and the layouts upon which it is based substantiate that these engines can be installed without compromising the orbiter's aft-fuselage configuration.

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Apogee

V = 5150 FPS
H = 202,500 Ft
t = 206 Sec

Onset Aero Forces

V = 5310 FPS
H = 173,000 Ft
t = 248 Sec
q = 20 PSF

Maximum Dynamic Pressure

V = 3760 FPS
H = 46,000 Ft
t = 306 Sec
q = 3070 PSF

Staging

V = 5502 FPS
H = 151,400 Ft
t = 146 Sec

Deploy Parachutes

V = 748 FPS
H = 12,430 Ft
t = 342.5 Sec

Impact

V = 148 FPS
H = 0
t = 398.4 Sec

17°

—172 N Mi—

Control Mode ——— RCS ——— Aero Surfaces —

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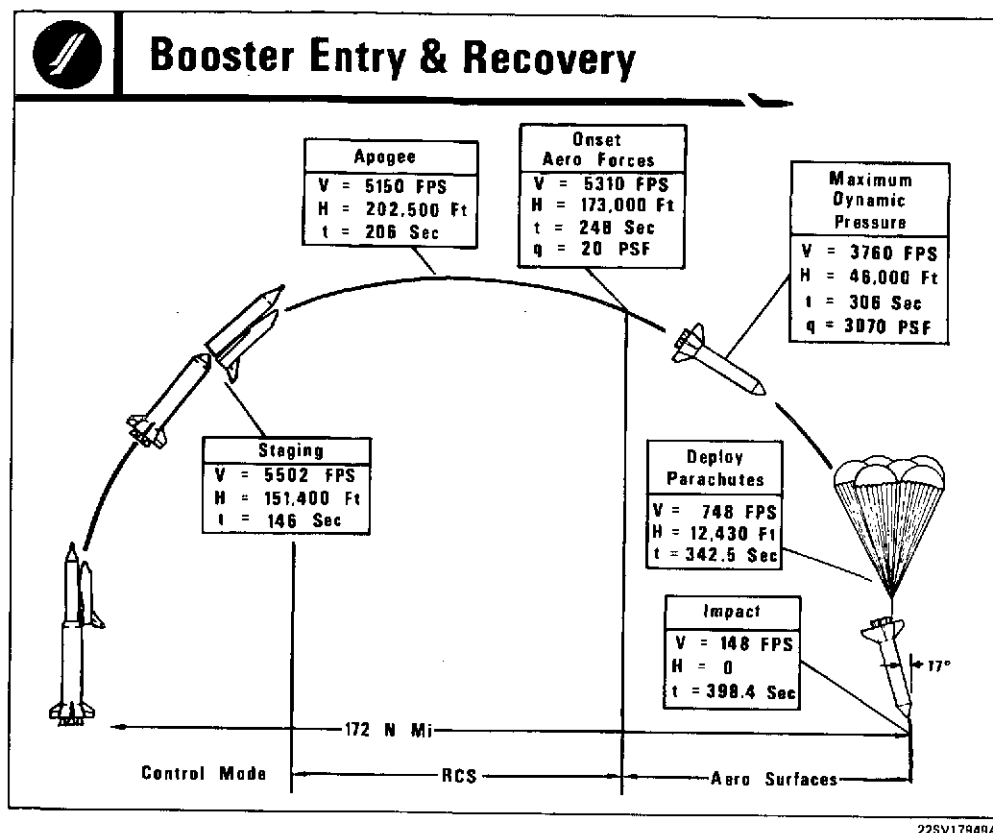
The primary booster issues are defined in four basic areas and are:

1. Pressure-fed engine and system development

    Weight and I_{sp}
    Combustion stability
    Pressurization

2. Entry techniques and requirements

    Stability and control

3. Recovery

    Drag level (body and flaps)
    Chute deployment
    Impact loads

4. Retrieval and Refurbishment

    Turnaround time/spares

The mission profile defines the critical elements of the flight
from launch, to staging, apogee, maximum dynamic pressure,
deployment, and impact. The chart illustrates the mission profile
elements and related issues.

Pressure-Fed Booster   B-19B-6

171.33 Ft

45° Nose Angle

40°

Fin (4)
446 Ft

15° Flare
Angle

73 Ft

Flap Control
Angles

±5°

35°

¢ Veh

5°

Expendable Tank
Interface

29.33 Ft
Dia.

275 in. Dia

(4) Hold
Down Fittings

45 Ft
Dia

Parachute
Stowage

Engines
1.17 M̄ Lb
SL Thrust

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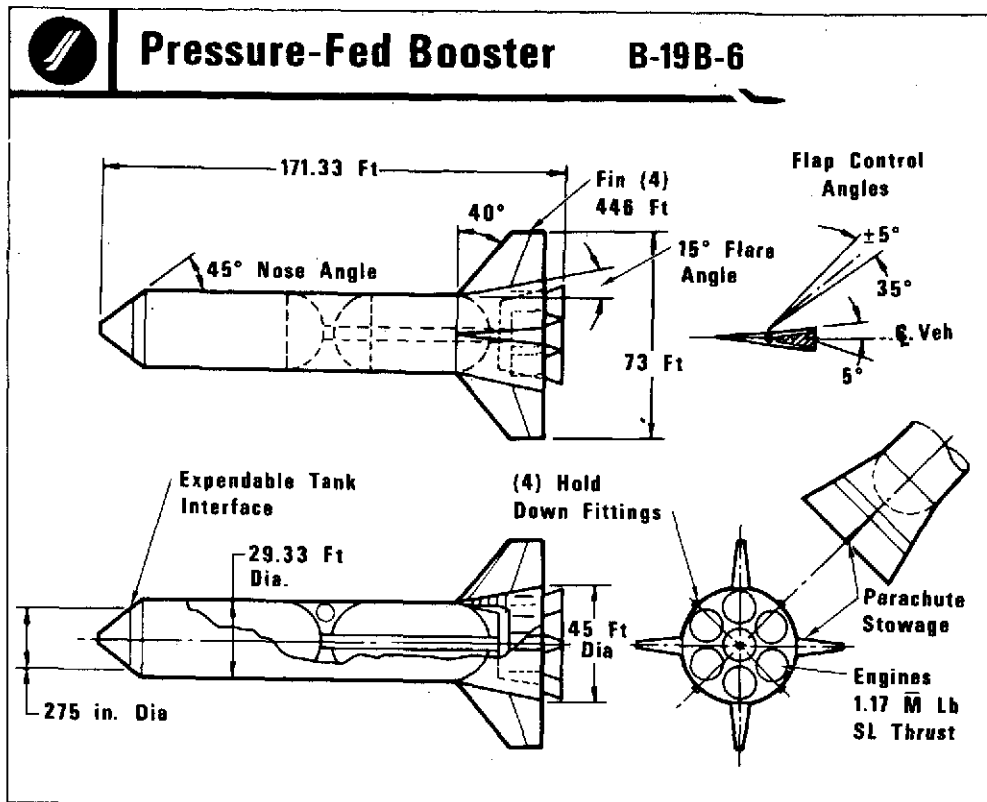
The pressure-fed booster is a reusable vehicle configured for a tandem arrangement with the orbiter and its external oxygen-hydrogen tank. The vehicle system is a series-burn type featuring a BLOW of 4,496,000 pounds, a staging velocity of 4800 fps, and a subsonically deployed parachute recovery system for controlling the impact to 150 fps with a recovery weight of 655,000 pounds.

The booster arrangement consists of a nose element, a forward LO_2 tank of 718 inconel, intertank, aft RP.1 fuel tank, and a thrust structure of 6A1-4V titanium. Four fins are provided with 718 inconel leading edges and flaps and titanium main box structure. Recovery parachutes are provided and stowed in the fin.

The main propulsion system uses seven pressure-fed engines, each rated at 1,043,000 pounds thrust (SL) with an LITVC system (liquid oxygen, 5-degree effective angle maximum). The propulsion pressurization uses pressurant LN_2/N_2H_4 to transfer propellants from the tanks to engines. The propellants are LO_2/RP.

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TRADE STUDY ITEM		STATUS
PROPULSION:	PROPELLANTS - O_2/C_3H_8 VS O_2/RP DUCT COOLING VS REGEN	SELECT: O_2/RP ΔBLOW = -230K LB REGEN ΔPROG COST = -$92M
	LITVC INJECTANT - LO_2	RETAIN (BUT STILL OPEN)
PRESSURIZATION:	PRESSURANT - LN_2/N_2H_4 VS VARIOUS ALTERNATIVES	RETAIN LN_2/N_2H_4
CONTROL:	ASCENT - LITVC & FINS VS GIMBAL & NO FINS	OPEN ISSUE
	ENTRY - FINS	RETAIN FINS
	POST STAGING - RCS & FINS	FINS ONLY (V_S <5,000 FPS)
CONFIGURATION:	TANK DIAM - L/D = 5.6 VS LOWER L/D	OPEN ISSUE
TANK MATERIAL:	RP TANK 718 VS Ti	SELECT Ti ΔBLOW = -120K LB
ENTRY/RECOVERY MODE:	BALLISTIC (BODY + FIN-FLAP DRAG) VS HIGH α	RETAIN BALLISTIC
	SUPERSONIC DROGUE CHUTES DEPL	ELIMINATE ΔBLOW = -40K
RETRIEVAL:	TOW BACK VS BARGE	RETAIN TOW BACK
ATTRITION:	16 VEHICLES	CHANGE TO 5 VEHICLES - BASED ON ATTRITION STUDY

22SV18145

This chart presents the status and shows there are still unresolved issues or design selections to be made.

Recommended Changes to PFB Baseline

	BASELINE	RECOMMENDED CHANGE
PROPULSION		
FUEL	PROPANE	RP
COOLING	DUCT	REGEN
FUEL TANK PRESSURE	295 PSIG	345 PSIG
OXIDIZER TANK PRESSURE	291 PSIG	291 PSIG
WEIGHT PER ENGINE	12,772 LB	14,005 LB
CONFIGURATION		
VEHICLE SIZE	29.3 FT DIA X 164 FT LG	27.8 FT DIA X 158 FT LG
RECOVERY SYSTEM	6 X 100 FT DIA CHUTES AT M < 0.9 (DELETE SUPERSONIC DROGUE CHUTES)	
SIZING IMPACT (TYPICAL FOR V_S = 5500 FPS)		
BOOSTER WT EMPTY	0.675 M LB	0.679 M LB
BLOW	5.243 M LB	5.209 M LB
GLOW	6.570 M LB	6.534 M LB
TEST		
RECOVERY SYSTEM	ELIMINATE SUPERSONIC CHUTE TESTING	

22SV18144

This chart indicates the recommended changes to the pressure-fed booster with the major issue being a shift to LO_2/RP propellant.

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F-1 Ballistic Recoverable Booster B-24B

33 FT DIA

OLOW	1,242 M LB
BLOW	4,032
GLOW	5,274
V_{STAGE}	5,890 FPS

(4) F-1 ENGS
1,685 M LB THRUST

15 x 60 FT
PL BAY

ENGINE
CLOSURE

|— 82.5 FT —|

96.3 FT

|—————— 244.3 FT ——————|
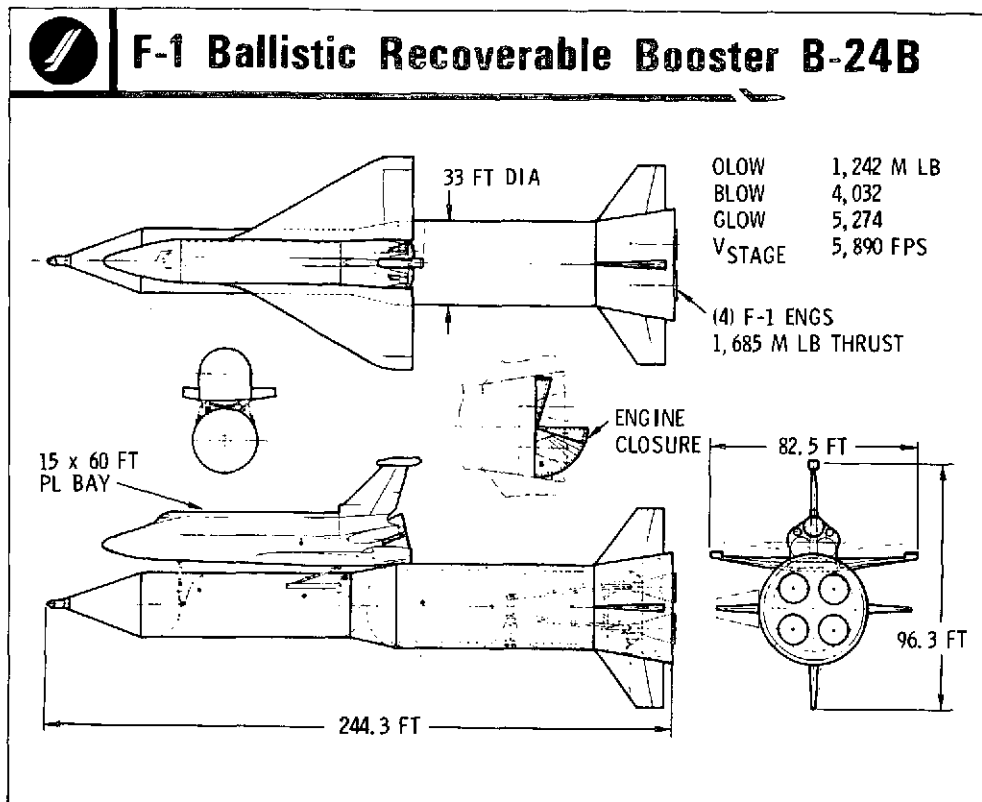
22SV18128

The pump-fed booster is a reusable vehicle configured for a tandem arrangement with the orbiter and its external oxygen-hydrogen tank. The vehicle system is a series-burn type featuring a BLOW of 4,187,000 pounds, a staging velocity of 5890 fps, and a subsonically deployed parachute recovery system. The recovery weight is 500,000 pounds.

The booster arrangement features sizing and configuration for commonality with Saturn S-IC to utilize existing technologies, tooling, and components. The aft end features an engine protection closure and four fins.

The main propulsion system uses five F-1 engines with gimballed nozzles. The propellants are LO_2/RP.

This section addresses three technical issues that have a significant impact on program costs.

Cost Driver Issues

ISSUE

- **Impact of EOHT Mass Fraction**
  (i.e., Monocoque VS Skin Stringer)

- **Weight Growth & Performance**
  Sensitivity

- **Orbiter Payload & 14 x 45 FT Payload**
  Bay Impact

22SV18237A

First of the three issues is how the external oxygen-hydrogen tank is constructed. Studies have shown that tanks of simple designs incorporating a monocoque structure can be fabricated at a low cost per pound. In general, however, they tend to be heavier for a given volume than skin-stringer construction. Skin-stringer tanks cost more per pound than monocoque tanks. The second issue is how growth margin and growth capability are designed into the system and the sensitivity of the three configurations under consideration to growth margin and capability and the cost impact. Finally, we discuss impact on the system cost of varying the up payload, the down payload, and the orbiter cargo bay size.

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EOHT

22SV18216

EOHT Structure Major Load Differences - Parallel vs Series

LH2 TANK
BENDING MOMENT REDUCED
BY BOOSTER LATERAL SUPPORT

LO2 INERTIAL
LOAD (3G) TAKEN
OUT INTO BOOSTER - LH2 TANK BY-PASSED
(PRELAUNCH TO BOOSTER MAX "G")

SUBSEQUENT TO
SOLID JETTISON
INERTIAL LOAD ON
85% FUEL TANK
IS 1.4G

EQUAL TO
ZERO

LO2 INERTIAL
LOAD (3G) &
BENDING MOMENT
TRANSFERRED THROUGH
LH2 TANK TO BOOSTER

PARALLEL

SERIES

22SV181256

In the parallel-burn systems, the moment introduced at the aft end of the tank is equal to 0. Further, the LOX inertial load is taken out of the external oxygen-hydrogen tank (EOHT) into the booster at the unpressurized interstage between the LOX and LH$_2$ tanks. The bending moment at this point is reduced by the lateral support provided by the booster. In contrast to this, the series-tandem arrangement introduces a significant bending moment into the aft end of the EOHT, and the LOX inertial load and bending moments must be transferred through the LH$_2$ tank into the booster. Thus, the load differences between the parallel- and series-burn systems should permit a lighter weight LH$_2$ tank to be designed for the parallel-burn, strap-on-booster arrangement. This is the case. It is noted, however, that the improvement in mass fraction for parallel burn is large for a monocoque tank but not nearly so prominent with a skin-stringer tank when these mass fractions are compared to those associated with a phantom series-burn arrangement.

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Impact of EOHT Design on Sys Parameters - PFB

GLOW (K Lb)

Monocoque

Skin Stringer

V_S (K FPS)

OLOW (K Lb)

Monocoque

Skin Stringe

V_S (K FPS)

Tank (K Lb)

Monocoque

Skin Stringer

V_S (K FPS)

Booster (K Lb)

Monocoque

Skin Stringer

•DESIGN POINT
10% MARGIN WITH
10% GROWTH CAP.

V_S (K FPS)

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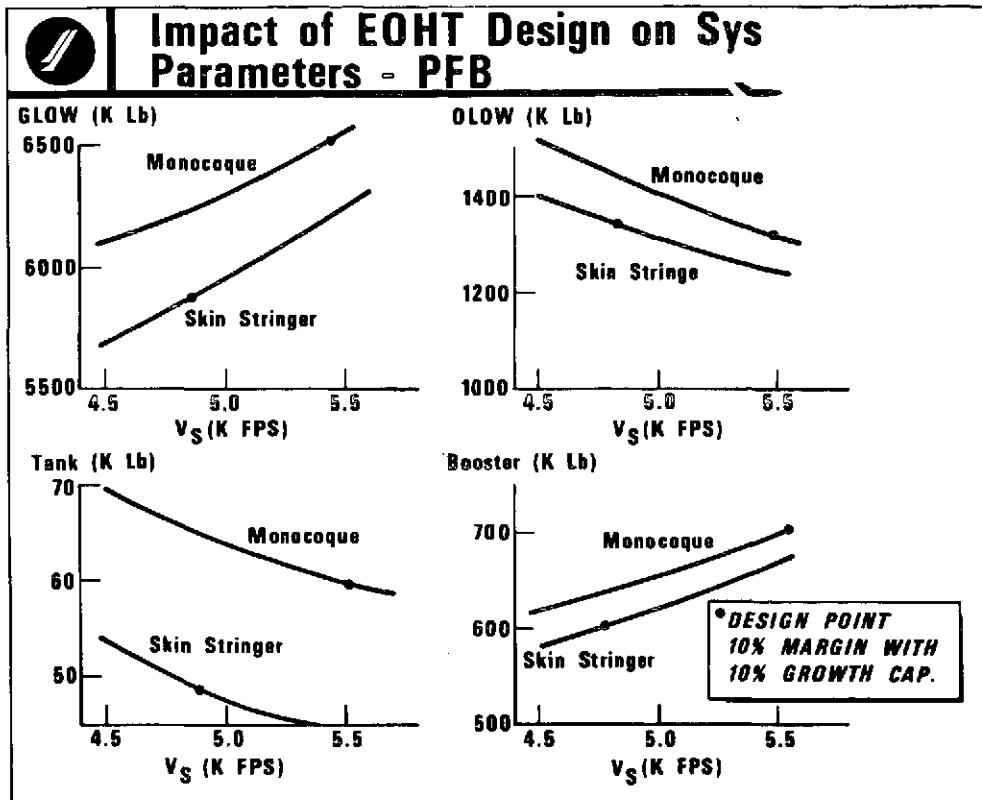
Variations of gross liftoff weight (GLOW), orbiter liftoff weight, empty external oxygen-hydrogen tank (EOHT) weight, and booster empty weight are plotted against staging velocity for two types of tank construction. Also shown on each curve is the design point selected to provide a 10-percent growth capability for both the booster and orbiter over the 10-percent initial margin design. A reduction in GLOW on the order of 600,000 pounds is obtained from switching from a monocoque to a skin-stringer tank. At the same time, a reduction in tank weight of approximately 11,000 pounds is obtained. Finally, booster empty weight is reduced approximately 100,000 pounds, which will be reflected in the design of the booster recovery system.

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Δ Total
Program
Cost $\overline{M}$

```
• Skin-Stringer EOHT
• Lower Program & Lower DDT&E Cost
```

Mono

Δ(DDT&E) = + 50\overline{M}$

Skin Stringer-Frame

Capability for 10%
Additional Wt Growth
(Booster & Orbiter)

V_S (K FPS)
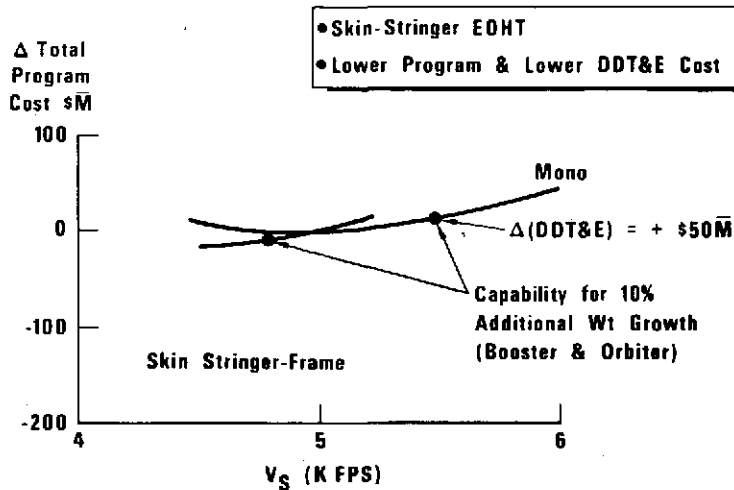
225V180566

Total program cost and the difference of the cost for design, develop-
ment, test, and evaluation (DDT&E) for two external oxygen-hydrogen
tank (EOHT) designs is plotted against staging velocity. Near the
design points where capability for 10-percent growth is available the
total program cost and the DDT&E cost are almost the same. The
lower staging velocity available with the skin-stringer tank will
significantly simplify recovery of the pressure-fed-booster (PFB).
Skin-stringer construction for the EOHT was selected as the base-
line for the pressure-fed-booster, series-burn, tandem-arrangement
system.

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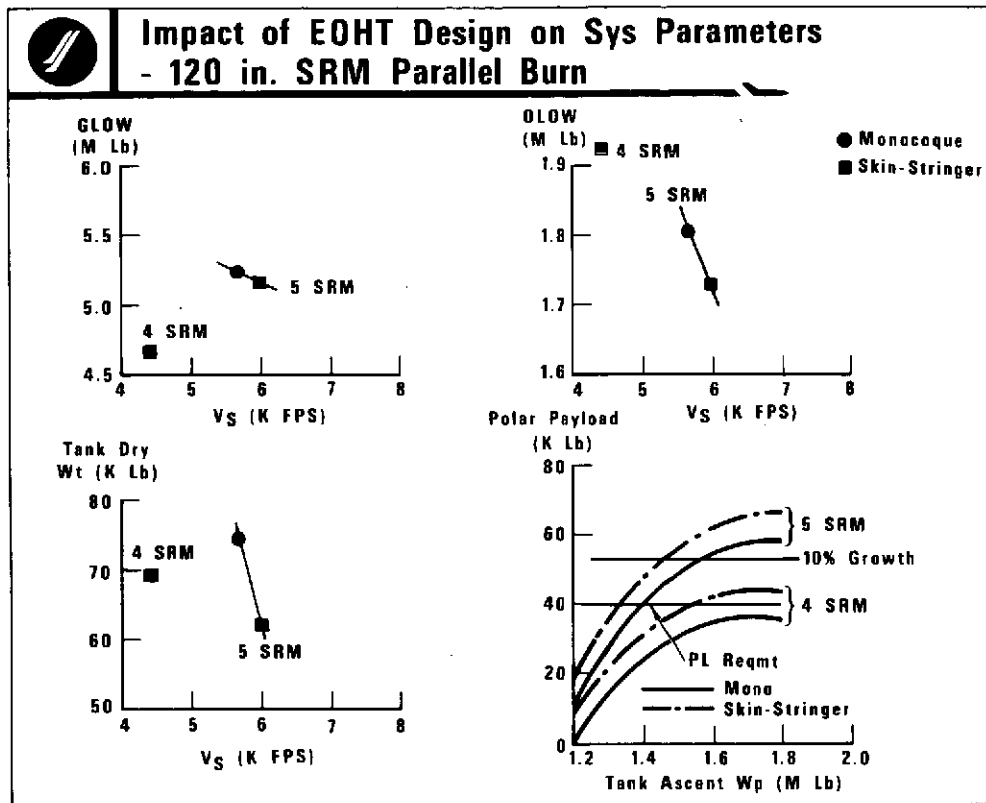
Shown are the variations in gross liftoff weight, orbiter liftoff weight, tank dry weight, and payload to polar orbit for a 120-inch solid-rocket-motor (SRM) parallel-burn system and two modes of tank construction. For a given number of SRM's, the booster weight is a constant. It is seen that regardless of tank construction, five SRM's are required to provide a 10-percent growth. With four SRM's, only skin-stringer construction will meet the minimum payload requirement. With five SRM's, the skin-stringer construction will save approximately 13,000 pounds of tank weight compared to monocoque construction.

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Impact of EOHT Design on Sys Cost - 120 in. SRM
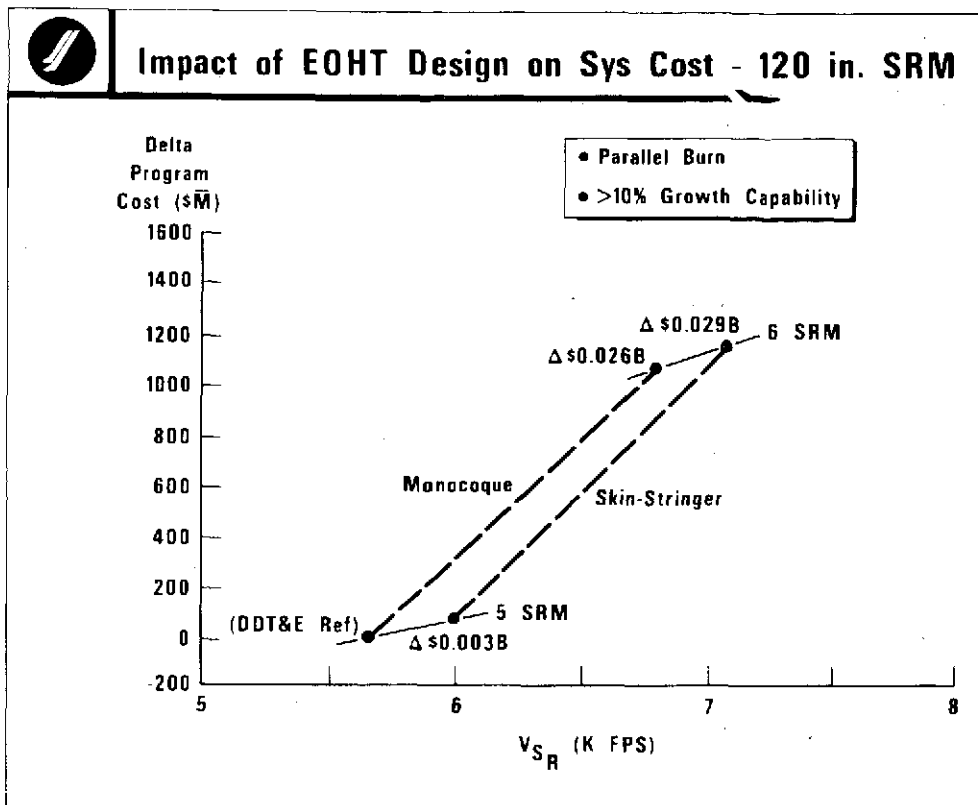
Shown in the chart is the impact on system cost of two methods of tank construction. Again it is seen that little difference in program cost or cost of design, development, test, and evaluation results from changing tank weight. However, in the solid-rocket-motor (SRM) system little advantage is gained from a skin-stringer tank because the booster weight is held constant for this type of system. Because of its simplicity and because recovery is not a consideration, it was elected to retain the monocoque construction for the 120-inch parallel-burn system. No advantage is gained with six SRM's, although significantly greater growth capability would be available. On this basis, further consideration of six SRM's was dropped.

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Impact of EOHT Design on Sys Parameters

156 in. SRM
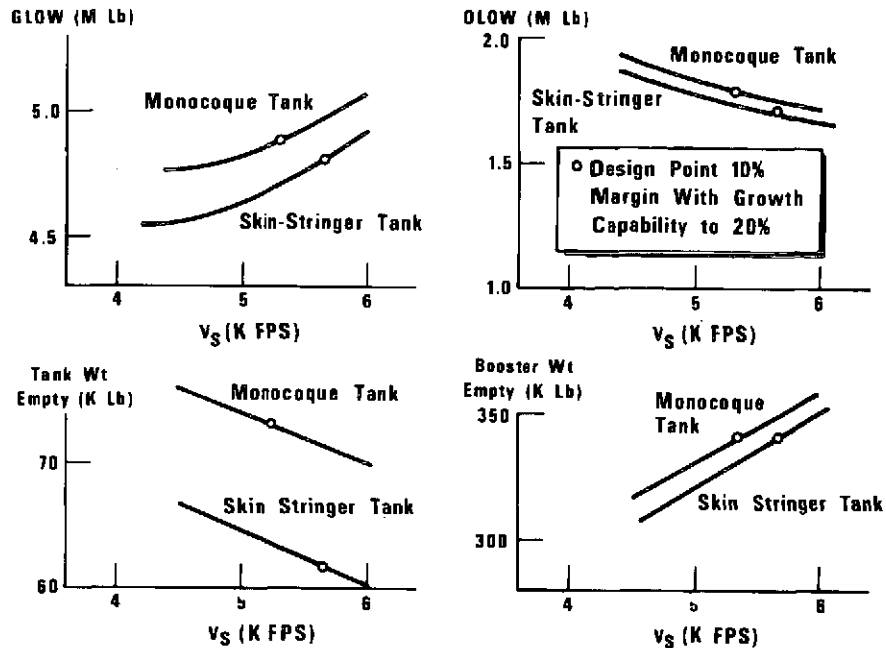
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The low orbiter-liftoff-weight empty tank weight and booster empty
weight are shown in the accompanying chart. Illustrated on each
curve are design points with a 10-percent margin and an additional
10-percent growth capability. Again it is seen that the skin-stringer
tank is significantly lighter than the monocoque tank at the design
points and saves approximately 11,000 pounds. However, no great
savings in booster empty weight results from changing the tank
construction.

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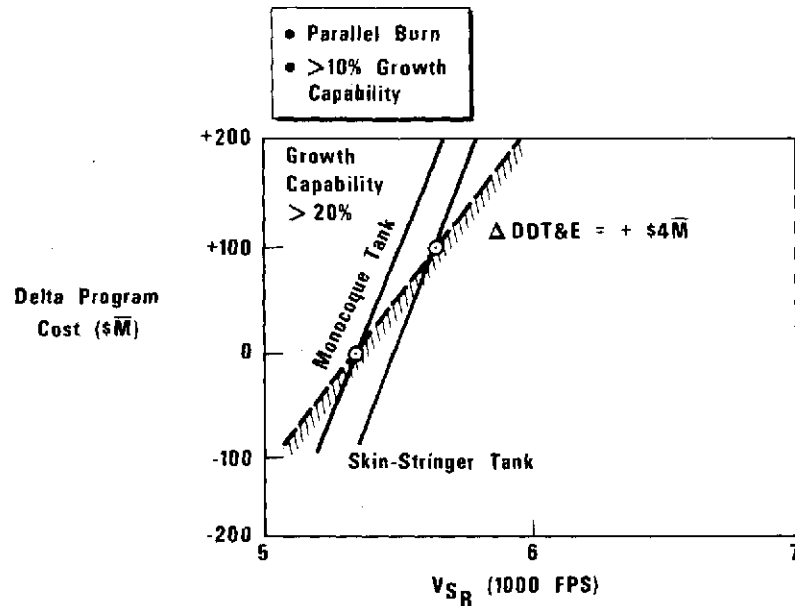
22SV1B1268

At the design points, it is seen that the monocoque tank would be
approximately one hundred million dollars cheaper in program cost
and about four million dollars cheaper in cost of design, development,
test, and evaluation on a total-system basis. Therefore, a monocoque
tank was selected for this parallel-burn system.

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Cost Driver Issues

ISSUE	FINDINGS
• Impact of EOHT Mass Fraction (i.e., Monocoque VS Skin Stringer)	• Skin Stringer Frame for Series Burn Systems (PFB & SRM'S) (Lightweight Monocoque for Parallel Burn Systems (SRM'S)
• Weight Growth & Performance Sensitivity	
• Orbiter Payload & 14 x 45 FT Payload Bay Impact	

22SV18238A

In summary, the impact of the external oxygen-hydrogen tank mass fraction on series-burn systems was seen to be of some advantage, particularly in recovery of the pressure-fed-booster. Although the programmatic costs were essentially equivalent to those resulting from use of a monocoque tank, the lightweight monocoque tank was more cost effective in the parallel-burn systems.

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GROWTH CAPABILITY

Historically, all vehicle systems gain in weight and degrade in performance from the time of authority to proceed through the first flight. Thus, two issues arise. The first is system design margins to be accounted for at program initiation together with the growth capability to build into the system. The second issue is the relative impact on system cost among the three systems being considered in this study.

- **OVERSIZE INITIALLY FOR ALL EXPECTED WEIGHT GROWTH**

    - Launch Partially Filled Tank Max Dynamic Pressure Constraint

    - Excess Payload Capability if Growth Fails to Develop

- **NO GROWTH PROVISIONS**

    - Expensive Redesign & Program Cost Escalation

- **INITIAL 10% MARGIN WITH GROWTH CAPABILITY TO 20% BY OVERSIZING BOOSTER**

    - If Growth Develops in Excess of 10% Resize Tank for Growth Up to 20%

22SV17886A

Weight histories from various programs have indicated that we can expect up to a 20-percent increase from go-ahead through first flight. Three options are available to account for this anticipated gain. First, the design can be oversized for the entire 20-percent gain expected. If the weight increase failed to develop, extra payload capability would be available.

A second option would be to provide no growth provisions. This would result in redesign of the vehicle propellant tanks and perhaps an upgrading of the booster engine thrust, all of which would mean costly design changes and schedule slips. A third option would be to incorporate a 10-percent margin in the initial design and provide capability to grow another 10 percent by resizing the external oxygen-hydrogen tank (EOHT). It would be anticipated that initial sizing would take place at the preliminary requirements review and the weight growth during the design would be monitored through preliminary design review, at which time the EOHT would be resized to gain back at least a 10-percent margin.

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Wt Growth Approach-Cost Comparison-PFB Series

Δ TOTAL PROGRAM COST
($MILLIONS)

INITIAL CAPABILITY
OF WT GROWTH TO 20%

MINIMUM COST SYSTEM
GROWTH NOT REQUIRED

EMPTY WEIGHT GROWTH MARGIN (BOOSTER & ORBITER)
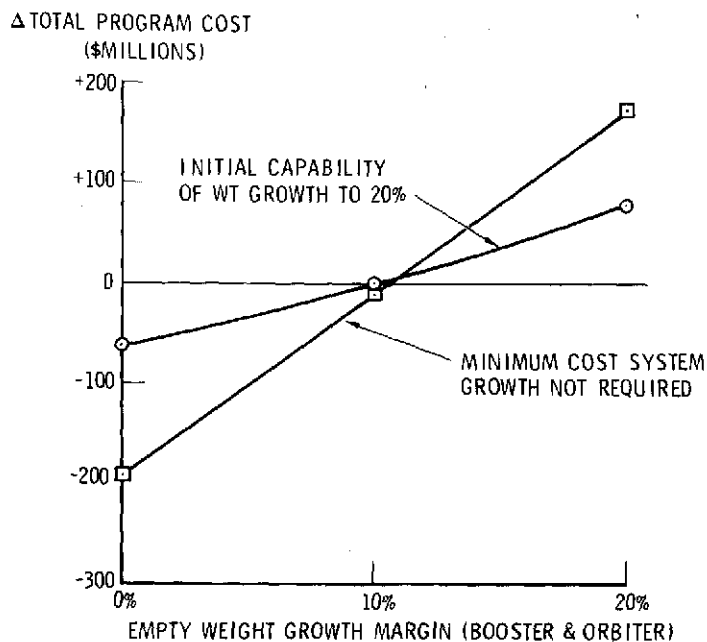
22SV18223A

Total program cost was evaluated as a function of the empty-weight growth margin for the combined booster and orbiter system for two cases: (a) where additional growth over the selected built-in empty weight growth margin was not required and (b) where capability to absorb weight growth up to 20 percent was required. These are shown in the accompanying chart. It is seen that if a 20-percent margin for weight increase is ultimately required, it is more cost effective to design in the capability to grow 20 percent. The reason is that, if a system is initially oversized for an empty-weight margin of 20 percent, then the orbiter wing is designed to meet the landing speed requirements for the extra weight. Where the system is designed for a lesser empty-weight growth margin, but the capability to grow to 20 percent, then the orbiter wing would be sized to accommodate the design landing speed at the initial weight growth margin. A deviation from this requirement would be accepted if the system grew 20 percent.

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Wt Growth Approach - Cost Comparison - 120 in. SRM

DELTA PROGRAM COST ($M)

- PARALLEL BURN
- 15 x 60 ORBITER, 40 UP/40 DN

GROWTH CAPABILITY TO 20%

5 x 1207 MONO

5 x 1207 MONO

5 x 1207 MONO

NO GROWTH CAPABILITY REQMT

STEP TO 5 SRM

4 x 1207 SKIN-STRINGER

4 x 1207 MONO

WEIGHT MARGIN*

*APPLIED TO ORB DRY WT, NEW 1207 STRUCT., 2% MAX ON TANK

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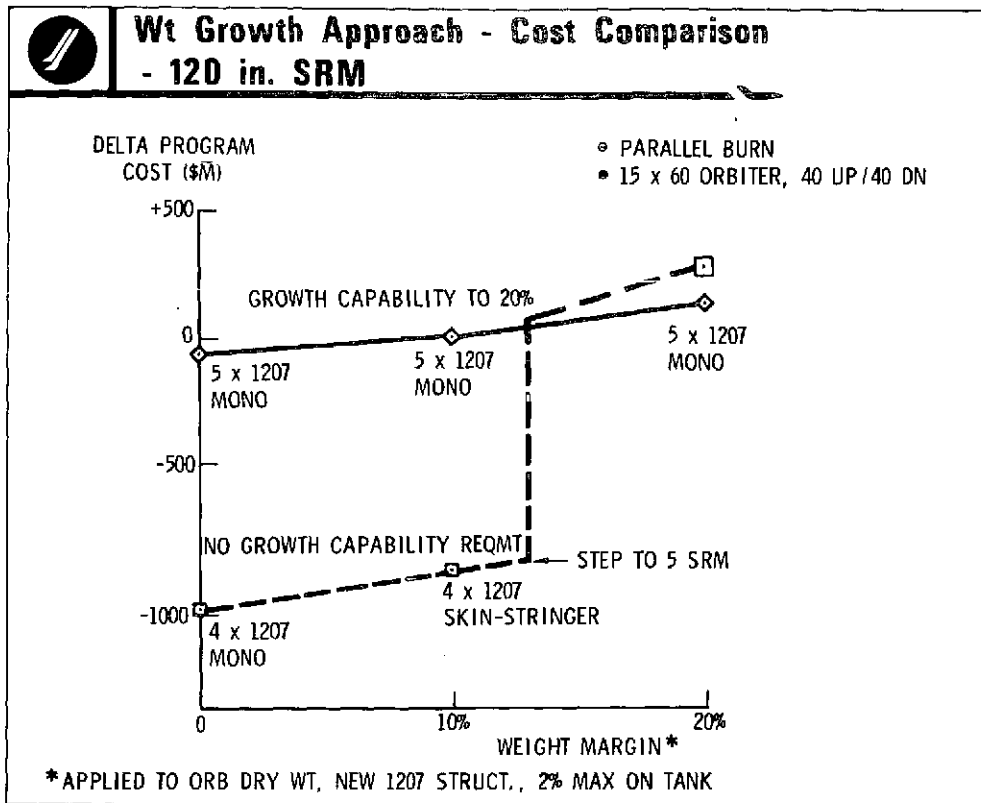
Delta program costs were calculated as a function of empty-weight margin for the 120-inch-solid-rocket-motor (SRM) parallel-burn systems for varying numbers of SRM motors. Where no specific growth requirement existed at up to about a 12-percent empty-weight margin, a four-engine SRM could meet the payload requirements. At that point, however, a step to five SRM's would be required. Note that the external oxygen-hydrogen tank construction has to be changed for a 10-percent margin to skin stringer in order to hold the SRM's to four. It is somewhat more cost effective to design in growth capability to 20 percent than to design in the entire 20-percent initially.

- 100 -

Wt Growth Approach - Cost Comparison - 156 in. SRM

DELTA PROGRAM COST ($M)

- PARALLEL BURN
- MONOCOQUE TANK

GROWTH CAPABILITY 20%

NO GROWTH CAPABILITY

WEIGHT MARGIN*

*APPLIED TO ORB DRY WT, INTERSTAGE STRUCT, 2% MAX ON TANK, 10% MAX ON SRM CASE

22SV18127A
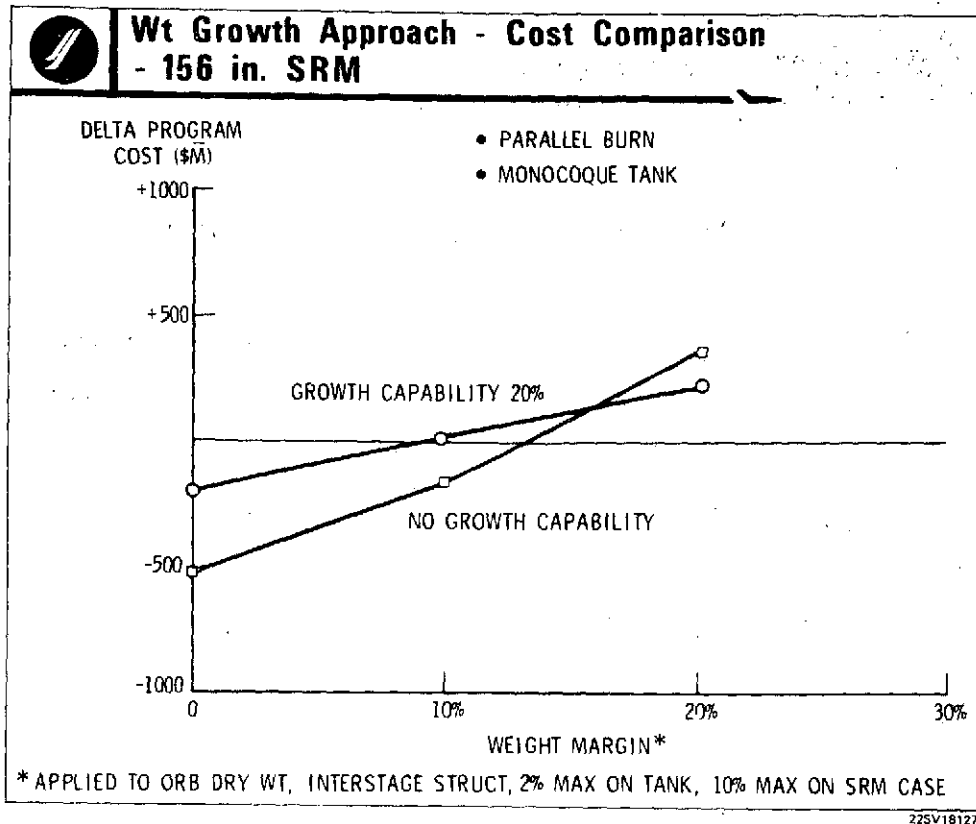
Delta program costs were calculated as a function of empty-weight margin for parallel-burn 156-inch solid-rocket-motor systems. The same result is seen here as in the previous two systems. It is more cost effective to design a system at a lower weight margin but with growth capability to 20 percent than it is to design in the entire 20 percent initially.

Cost Driver Issues

ISSUE	FINDINGS
• Impact of EOHT Mass Fraction (i.e., Monocoque VS Skin Stringer)	• Skin Stringer Frame for Series Burn Systems (PFB & SRM'S) (Lightweight Monocoque for Parallel Burn Systems (SRM'S)
• Weight Growth & Performance Sensitivity	• Not a Discriminator. All Systems Comparable
• Orbiter Payload & 14 x 45 FT Payload Bay Impact	

22SV18239A

In all three systems it appeared to be cost effective to design in a weight-growth margin of 10 percent with capability to grow another 10 percent during the program. Calculated performance partials for all systems were approximately the same.

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PAYLOAD

&

CARGO BAY SIZE

An investigation was made to determine the impact of up payload, down payload, and payload bay size on the total system costs. This involved investigation of the orbiter configuration and its aerodynamic characteristics and subsequently synthesizing and costing various launch systems with the various payloads and orbiters.

Effect of Down P/L Wt & P/L Baysize on Orbiter Wt

Item	Cargo Bay		
	15 Ft × 60 Ft		14 Ft × 45 Ft
	65/40	45/25	45/25
Wing	14,023	12,475	11,702
Tail	3,436	3,436	3,436
Body	34,075	34,075	32,248
TPS	26,326	25,084	22,788
Landing, Dock	12,297	11,338	11,039
Propulsion, ASC	23,613	23,613	23,613
Propulsion, Crus	114	114	114
Propulsion, Aux	5,483	5,483	5,262
Prime Power	3,492	3,435	3,418
Elec Conv & Dist	3,668	3,668	3,668
Hyd Conv & Dist	1,376	1,376	1,376
Surface Control	1,745	1,631	1,596
Avionics	3,982	3,982	3,982
Environ Cont	2,560	2,560	2,560
Personnel Prov	1,617	1,617	1,617
Growth	12,088	11,697	11,149
Total (Dry)	149,895	145,584	139,566

22SV180800

The orbiter vehicles were synthesized, assuming various down payloads and cargo bay sizes. A comparison of these is shown in the accompanying chart. As the down payload decreases from 40,000 to 25,000 pounds, the wing size also changes, the related thermal protection system decreases, the landing gear system is scaled as a function of the total system weight and the requirements for aerodynamic surface control decrease. The net change in oribter dry weight for a decrease in down payload from 40,000 to 25,000 pounds was approximately 4300 pounds. When the cargo bay length and diameter were changed to 14 by 45 feet, also with a 25,000-pound down payload, the fuselage shortened as shown and the wing sized decreased, and there was an associated decrease in thermal protection system weight. Again, the landing gear was scaled with the vehicle weight, as were the surface control requirements. Thus the net decrease in total orbiter weight from the baseline system was determined to be approximately 10,000 pounds.

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Orbiter Comparison Effect of Payload Bay

	Payload Bay	
	15 × 60	14 × 45
Wing Area (Sq Ft)	3,440	2,962
Fwd c.g. in from Nose	876	770
Aft c.g. in from Nose	911	788
Payload (Lb)	40,000	25,000
Landing Wt (Lb)	194,300	188,400
Aero Characteristics		
Entry Characteristics		
α (Deg)	31	29
L/D	1.4	1.4
$W/C_L S$ (PSF)	92.5	109
Max α (Deg)	50	50
Low Speed Characteristics		
L/D (Trimmed)	5.65	5.40
Nom c.g. in from Nose	895	778
Min Static Margin %L	0.5	0.5
Des TD Speed Knots	156	156

Modifications from 040A
* Increase Forebody Width 45 in.
* Reduce Wing Sweep from 60 to 50°
* Increase Wing Area 10%

22SV18118D

A pictorial comparison between the baseline (15 by 60 feet) and the smaller (14 by 45 feet) cargo bay orbiters is presented in this chart. The orbiter geometries, hypersonic entry characteristics, and low-speed landing characteristics are also compared.

Both orbiters include modifications made to the -040A baseline. They include:

1. Increased forebody width
2. Decreased forebody camber
3. Hard chine radius on forebody
4. Change in body length
5. Decreased wing sweep from 60 to 50 degrees
6. Resized wings to provide minimum touchdown speed $V_{TD} = 156$ knots with zero static margin at the aft center of gravity.
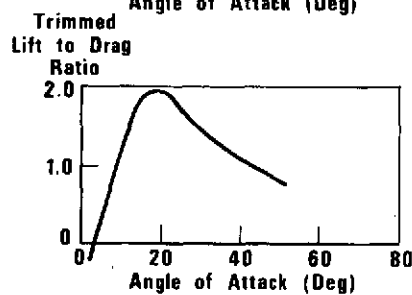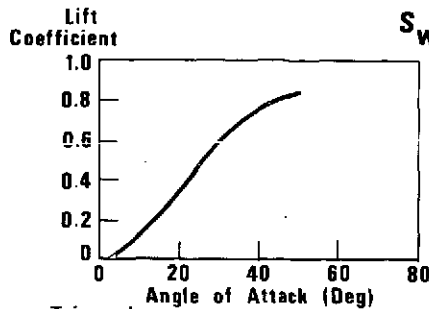
The above configuration changes provided a balanced hypersonic/ subsonic trim and balance capability.
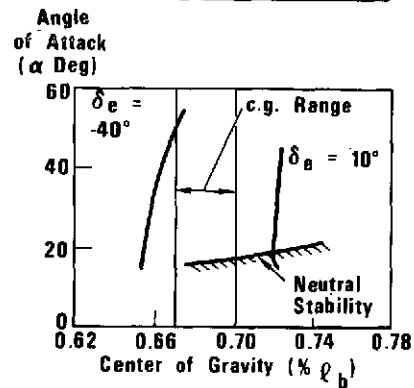
15 x 60 Cargo Bay

$$S_W = 3440 \ Ft^2$$

Lift Coefficient

Entry Characteristics		Value
L/D		1.4
W/C$_L$S	PSF	92.5
α	Deg	31

Angle of Attack (α Deg)

Trimmed Lift to Drag Ratio

Angle of Attack (Deg)

$\delta_e = -40°$ c.g. Range $\delta_e = 10°$

Neutral Stability

Center of Gravity (% ℓ_b)

Data Base: LRC 31 In. CFHT Test No. 80

22SV17916A

Hypersonic longitudinal aerodynamic estimates were made by using the Newtonian flow theory. The estimates were adjusted by employing correlated wind tunnel results from previous hypersonic wind tunnel tests for similar configurations.

The increased forebody width has provided adequate hypersonic trim capability at the forward (67-percent ℓ_b)and aft (70-percent ℓ_b) center-of-gravity positions. Elevon effectiveness data were obtained from the Langley Research Center CHFT Test 80.

The maximum lift-to-drag (L/D) ratio is 1.9, and the angle of attack for L/D = 1.4 is 31 degrees. The corresponding lift coefficient is 0.61 at 31-degree angle of attack which results in a trajectory parameter $W/C_L S$ = 92.5 psf.

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Subsonic Longitudinal Characteristics

15 x 60 Cargo Bay
$S_w = 3440$ Ft^2

Lift Coefficient

$\delta_e = 0°$

10°

Dashed: Trimmed ~ 0.68 ℓ_b

Angle of Attack (Deg)

Lift Coefficient

$\delta_e = -10°$ $\alpha = 20°$

$\alpha = 15°$

$\alpha = 10°$

$\alpha = 5°$

$\delta_e = 0°$

$\alpha = 0°$

Pitching Moment (C_m 0.68ℓ_b)

Trimmed Lift to Drag Ratio,

Angle of Attack (Deg)

Angle of Attack (Deg)

AC

c.g. Range

$\delta_e = -10°$

Center of Gravity (%ℓ_b)
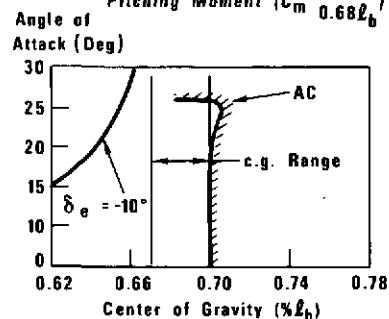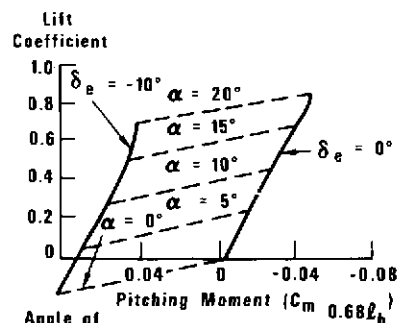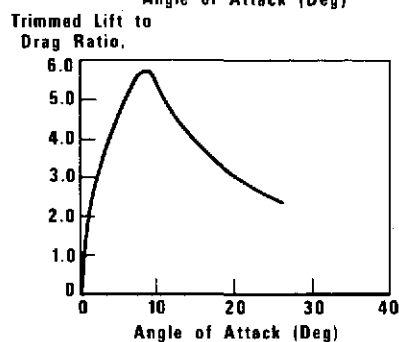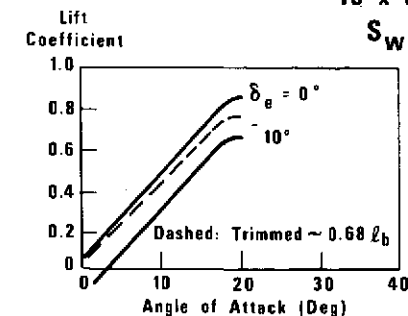
Data Base LRC-LTPT 85

22SV17915A

The data presented were based on wind tunnel results from the Langley Research Center LTPT Test 85 and North American Rockwell NAL Test 660, adjusted to reflect the following changes made to the 040A orbiter:

1. Increased forebody width
2. Decreased forebody camber
3. Hard chine radius on forebody
4. Increased body length
5. Decreased wing sweep from 60 to 50 degrees
6. Increased wing area
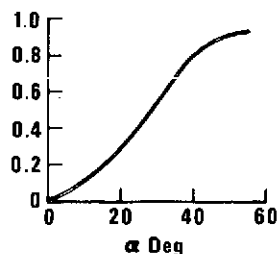
The vehicle exhibits static longitudinal stability across the angle-of-attack range with a 0-percent static margin at the aft center-of-gravity position (70-percent ℓ_b).

The trimmed lift coefficient at 17-degree angle of attack is 0.7 corresponding to -5.2-degree elevon deflection. The maximum trimmed lift-to-drag ratio is 5.65 at 8-degree angle of attack.
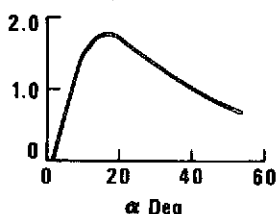
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Lift
Coefficient

14 Ft x 45 Ft Cargo Bay
S_W = 2962 Ft^2

Entry Chacteristics		Value
L/D		1.4
W/C$_L$S	(PSF)	109
α	(Deg)	29

1.0
0.8
0.6
0.4
0.2
0

0   20   40   60
α Deg

Trimmed L/D
2.0

1.0

0

0   20   40   60
α Deg

Trimmed Angle
of Attack
60   c.g. Range
                         δe = 0°
40   δe = +5°
                         δe = -20°
                    Neutral Stability
20

0
0.62 0.64 0.66 0.68 0.70
Center of Gravity (% ℓb)
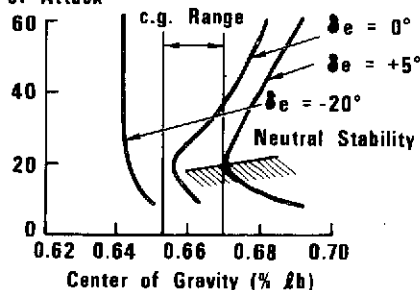
22SV17965B

Estimates for hypersonic longitudinal characteristics were made by
using Newtonian flow theory adjusted by correlated wind tunnel results
from previous hypersonic tests for similar configurations.

The increased forebody width has improved the trim capability substan-
tially at the forward (65.3%) and aft (67.7%) center-of-gravity positions,
over the previous 040A configuration. The shortened body has also
reduced the elevon deflections required to trim from -40 to -20 degrees
at the forward center-of-gravity and from +10 to +50 degrees at
the aft center-of-gravity position.

Elevon effectiveness data are based on wind tunnel data from the LRC
LTPT Test 80.

The maximum lift-to-drag (L/D) ratio is 1.7, and the angle of attack
for L/D = 1.4 is 29 degrees. The corresponding lift coefficient is
0.52 at 29-degree angle of attack, which results in a trajectory
parameter $W/C_L S$ - 109 psf.

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Subsonic Longitudinal Characteristics

22SV17964B
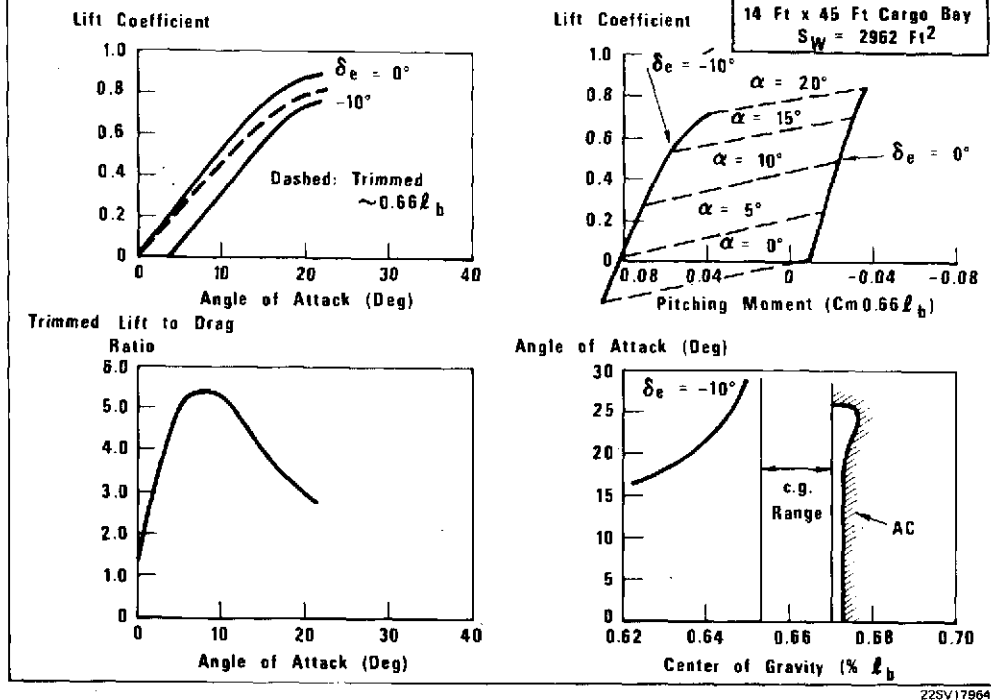
Estimates of the low-speed longitudinal characteristics for the short (14 by 45) cargo bay orbiter were based on wind tunnel data from the Langley Research Center LTPT Test 85 and the North American Rockwell NAL Test 660. The data were adjusted to reflect the changes made to the baseline 040A configuration:

1. Increased forebody width
2. Decreased forebody camber
3. Hard chine radius on forebody
4. Decreased body length
5. Decreased wing sweep from 60 to 50 degrees
6. Decreased wing area

Static longitudinal stability exists across the angle-of-attack ranges with a 0.3-percent static margin at the aft center-of-gravity position (67-percent $_b$).

The large elevons are very effective in providing low-speed trim. The trimmed centerline at 17-degree angle of attack is 0.7, requiring an elevon deflection of -6 degrees. The maximum trimmed lift-to-drag ratio is 5.4 at 7-degree angle of attack.

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22SV18098A
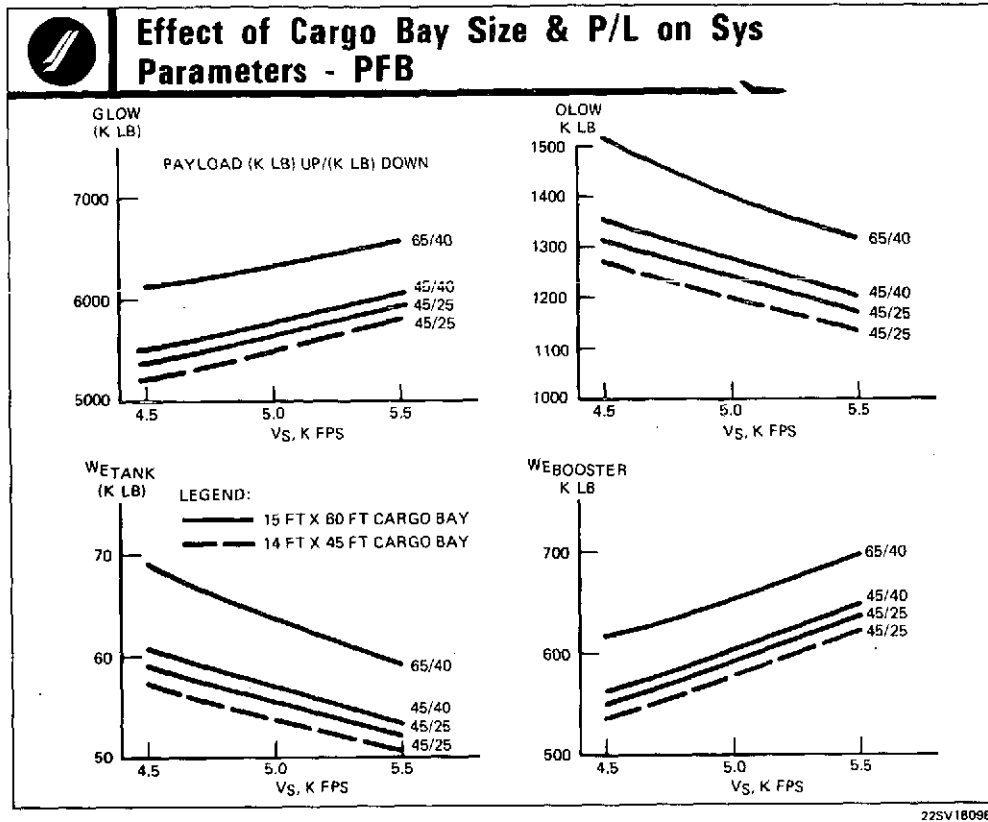
Vehicle parameters for a series burn pressure-fed booster (PFB) launch vehicle system were calculated. Gross liftoff weight (GLOW), orbiter liftoff weight (GLOW), empty tank weight, and empty booster weight are shown versus staging velocity. The major decrease in any of the system weight parameters is obtained by means of reducing the up payload. Subsequent decreases are small.

Effect of Cargo Bay Size & P/L on Cost-PFB

LEGEND:
——— 15 FT X 60 FT CARGO BAY
—— — 14 FT X 45 FT CARGO BAY
PAYLOAD (K LB) UP/K LB DOWN

PAYLOAD 65/40

10% ADDITIONAL WT GROWTH CAPABILITY (BOOSTER & ORBITER)

BASELINE (DDT&E)

$\Delta_T = \$160\overline{M}$

45/40

$\Delta_T = \$80\overline{M}$ 45/25

Δ(DDT&E) =   -$66\overline{M}$

Δ(DDT&E) =   -$28\overline{M}$

Δ(DDT&E) =   - 75$\overline{M}$

$\Delta_T = \$120\overline{M}$

45/25

TOTAL   =   -$169\overline{M}$
Δ(DDT&E)

ΔTOTAL = $360\overline{M}$

V_{STAGE} (K FPS)

Δ TOTAL PROGRAM COST ($\overline{M}$)
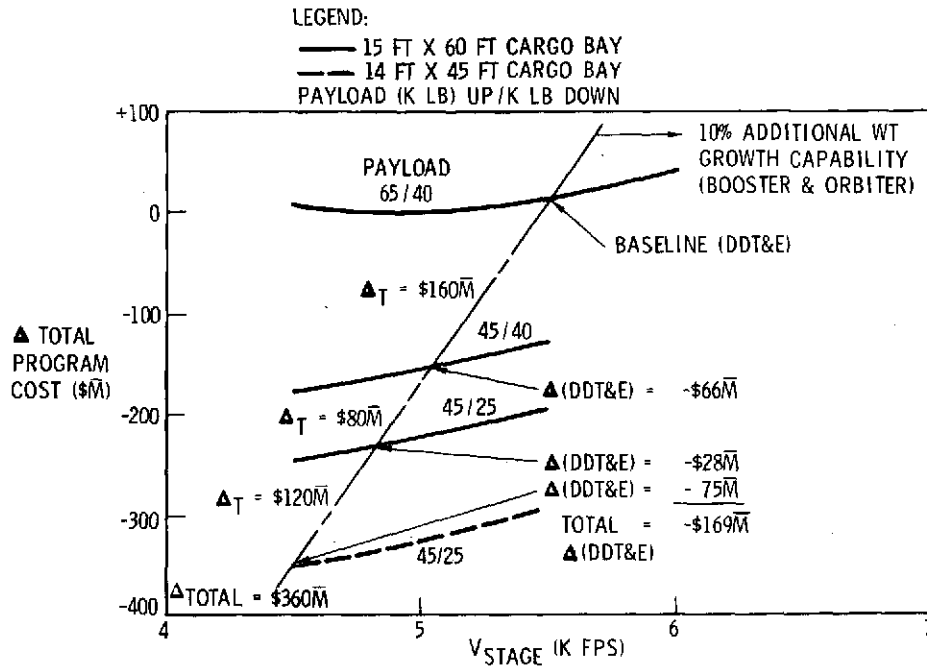
225V18095A

Total program costs and delta design, development, test, and evaluation (DDT&E) costs are shown for the four systems analyzed. As expected, the major decrease in program costs and DDT&E costs is derived from the decrease in up payload, subsequent cost benefits being derived from changes in down payload and cargo bay size.

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22SV18224A

System parameters were analyzed for the four orbiter configurations under consideration by using 120-inch solid rocket motors (SRM's) in a parallel burn mode. Combinations of four and five 120-inch SRM clusters were considered. When the up payload is reduced to 45,000 pounds, four SRM's are adequate to accomplish the mission regardless of the payload bay size or the down payload.

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Parallel Burn

Delta Program
Cost ($M̄)

```
  0  ┤─────────────────────────────────────  15 × 60
                                              65/40

                                              (DDT&E
                                               Ref)

-500 ┤

-1000┤         15 × 60      15 × 60
         14 × 45  45/25      45/40
         45/25
                (-35M̄)      (-34M̄)
         (-73M̄)
-1500┤
```

                         4 × 1207   5 × 1207
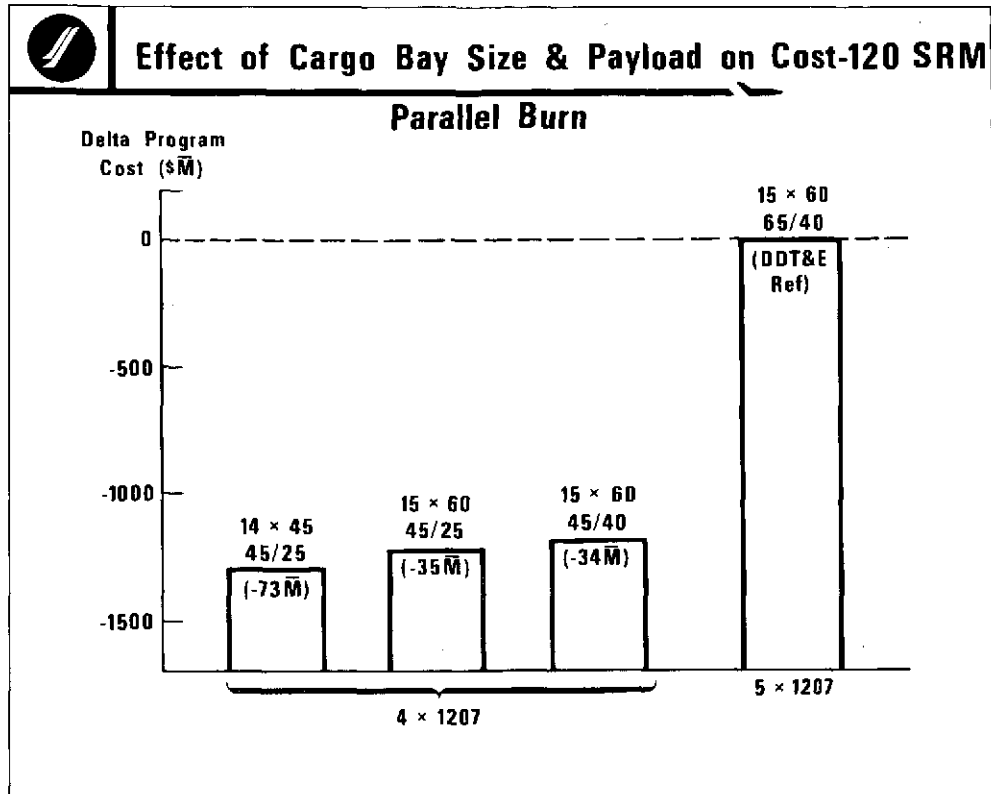
22SV18105B

Delta program costs were computed for the four systems under
consideration. Again, as with the pressure-fed system, the reduction
in up payload provides the major portion of the cost savings, much
smaller cost savings being accrued from changes in orbiter size.
This trend results from the fact that the boosters are discrete elements
and do not change in cost as orbiter propellant weight and payload
size changes.

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Effect of Cargo Bay Size & Payload on System Parameters-156 in. SRM
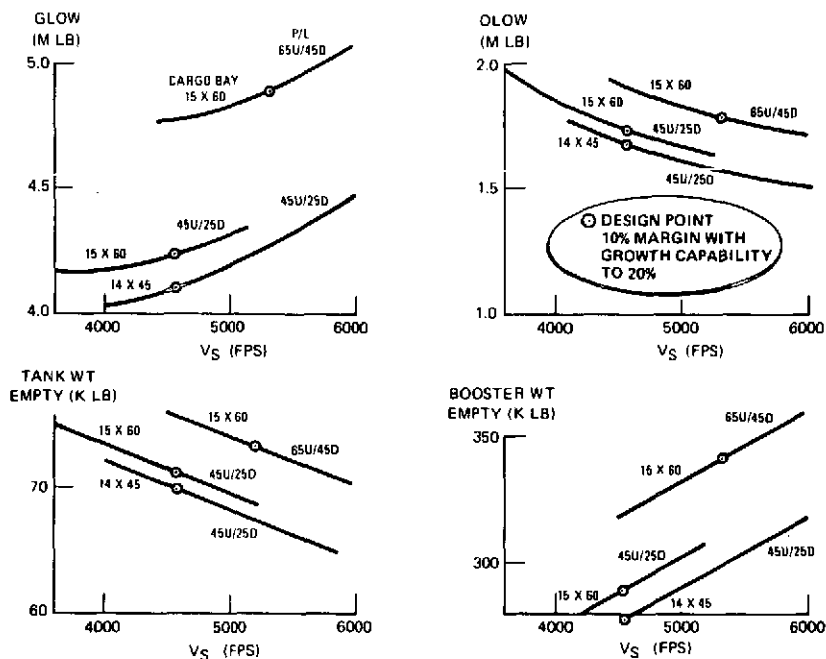
22SV18226A

System parameters were calculated in terms of gross liftoff weight (GLOW), orbiter liftoff weight (OLOW), empty tank weight, and booster empty tank weight versus staging velocity for the orbiter and payload combinations under consideration. The trend for this system remains the same as the trends for the 120-inch solid rocket motor (SRM) parallel burn systems and the pressure-fed-booster (PFB) series burn systems in that the major weight savings accrued from changes in up payload.

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Effects of Cargo Bay Size & P/L on Cost

156 in. SRM Parallel Burn
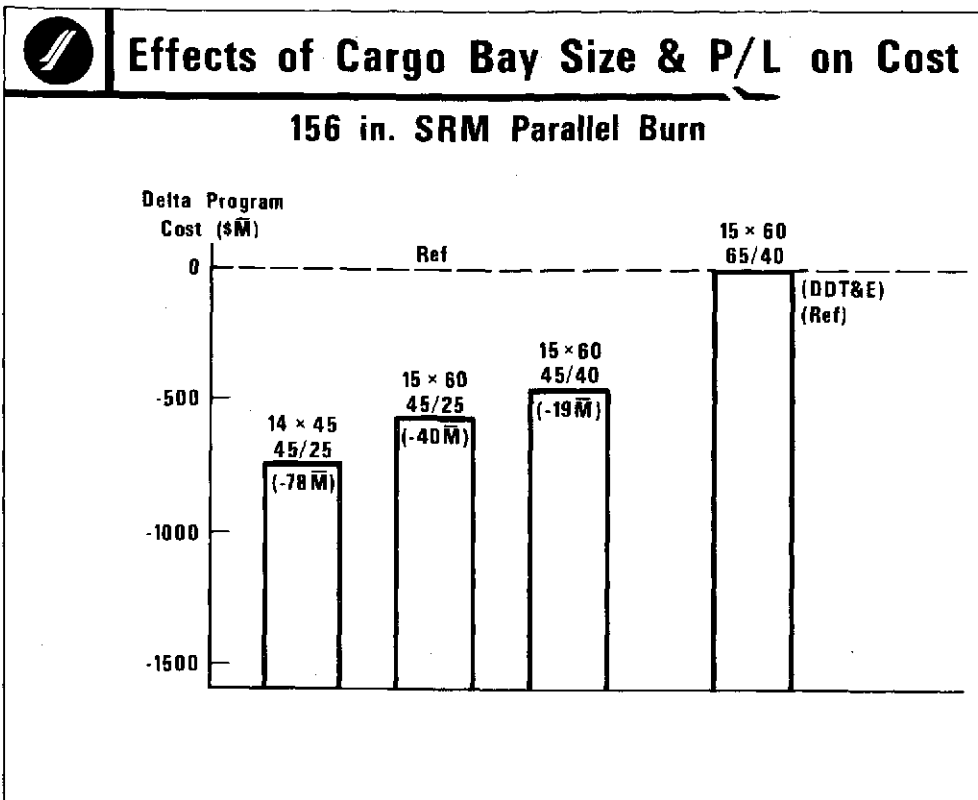
Delta program costs were calculated for the system design points
referenced to a baseline for a large cargo bay with full payload
carrying capability. Again, approximately half the cost savings
are accrued from reducing the up payload, the remainder being
accrued from reduction in orbiter size and down payload.

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System Comparison-PFB VS SRM Parallel

$/FLT	$7.6M	$7.25M	$13.3M	$11.9M

PFB SERIES

Total: 9.4B · 8.8B · 10.6B · 9.8B

DDT&E: 4.6B · 4.4B · 3.7B · 3.67B

Peak/Yr: 1.2B · 1.1B · 0.9B · 0.89B

15×60 (65U/45D)	14×45 (45U/25D)	15×60 (65U/45D)	14×45 (45U/25D)

156 IN. PARALLEL
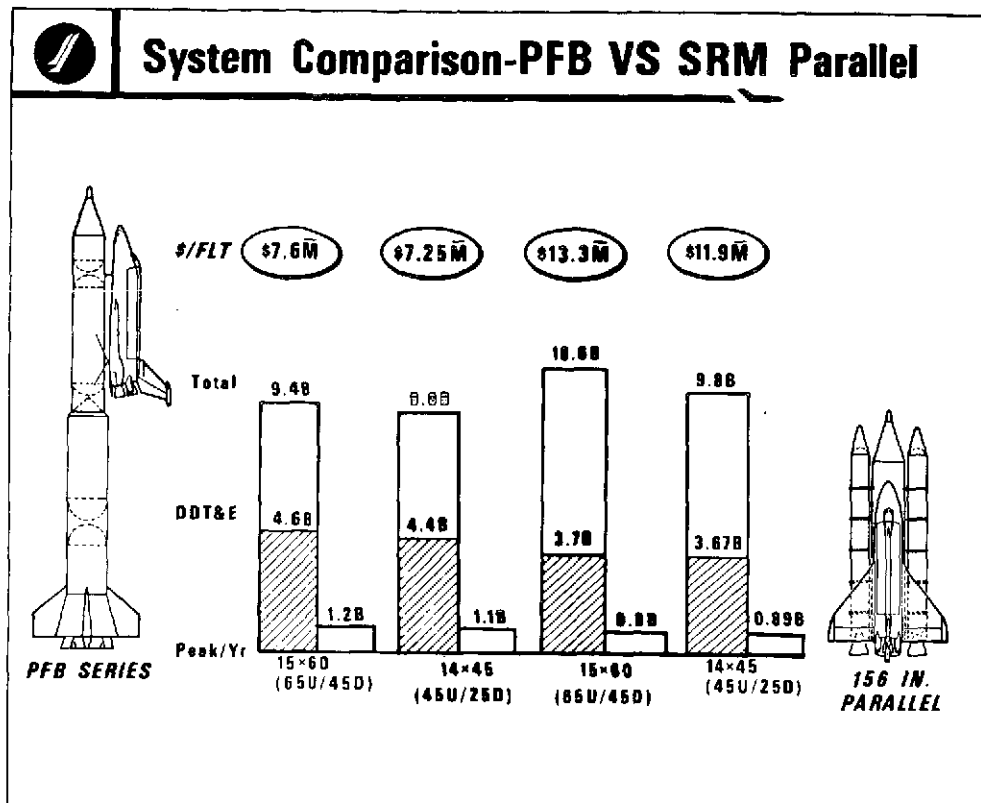
A direct comparison of specific interest is made between the pressure-fed-booster (PFB) series burn system with a large orbiter and design payloads or a small orbiter with reduced payloads and a 156-inch solid rocket motor (SRM) parallel burn system with the large orbiter and design payloads and a small orbiter with reduced payloads. As expected, the solid parallel system has higher cost per flight and higher program costs than the PFB system. The design, development, test, and evaluation (DDT&E) costs, however, are below the 4-1/2-billion-dollar target, and the peak-year annual funding is also below the cost target. For either system, however, the reduction in costs for the total program DDT&E or peak-year annual funding is not affected significantly by the orbiter size or up payload, but the cost per flight is significantly reduced for the parallel burn solid system.

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Cost Driver Issues

ISSUE	FINDINGS
• Impact of EOHT Mass Fraction (i.e., Monocoque VS Skin Stringer)	• Skin Stringer Frame for Series Burn Systems (PFB & SRM'S) (Lightweight Monocoque for Parallel Burn Systems (SRM'S)
• Weight Growth & Performance Sensitivity	• Not a Discriminator. All Systems Comparable
• Orbiter Payload & 14 x 45 FT Payload Bay Impact	• Major Cost Savings From Reduced Up-Payload, Secondary Savings From Reduced Down Payload & 14x45 Payload Bay

22SV18240A

Reduction in orbiter payload and in payload bay size has the following impact on the program costs: Major cost savings from reduced up payloads are accrued. Secondary savings from reduced down payload and a smaller orbiter are also available.

🖋 System Size Selection-PFB
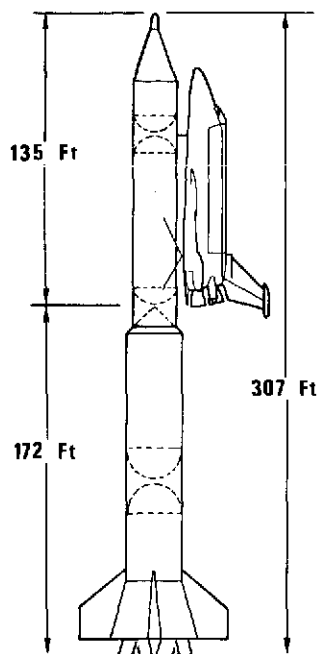
		Booster	Orbiter
Cargo Bay 15 × 60 Ft **Payload 40 K Polar**			
System GLOW	(K Lb)	5,842	
T/W at Liftoff		1.25	1.55
Stage GW	(K Lb)	4,496	1,345
Main Propellant	(K Lb)	3,767	975
OMS Propellant	(K Lb)		11.9
Stage Dry Wt	(K Lb)	603.7	149.9
Stage Landing Wt	(K Lb)		194.3
External Tank Wt	(K Lb)		48.7
$V_{Staging,}$	(FPS)	4,800	4,800
q_{Max}	(PSF)	647	647
$q_{Staging,}$	(PSF)	74.0	74.0
$\gamma_{Staging,}$	(Deg)	22.6	22.6
$\eta_{Staging,}$	(K Ft)	136.3	136.3
Engines		7 PF	3 Hi Pc
Thrust/Eng	(K Lb)	1,043 (SL)	472 (VAC)
I_{sp} - Sec (VAC)		273.4	456.3

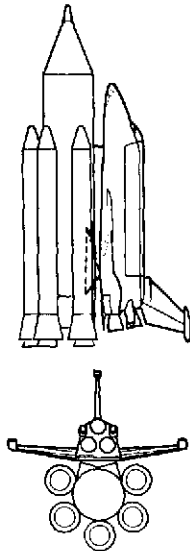Dimensions shown: 135 Ft, 172 Ft, 307 Ft

22SV18225B

The major system parameters for the selected pressure-fed booster series burn tandem arrangement system are shown. The external oxygen-hydrogen tank (EOHT) for this system features skin stringer type of construction.

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PARALLEL BURN

Cargo Bay Payload		15 Ft × 60 Ft 40K Lb Polar	
		Booster	Orbiter
System GLOW	(K Lb)	5,242	
T/W at Lift-off		1.35	
Stage Gross Wt	(K Lb)	3,438	1,804
Ascent Wp	(K Lb)	2,932	1,407
Stage Dry Wt	(K Lb)	506	150
Orbiter Land Wt	(K Lb)		194
External Tank			Monocoque
External Tank Wt	(K Lb)		75
q_{Max}	(PSF)	650	
V_S	(FPS)	5,670	
q_S	(PSF)	54	
γ_S	(Deg)	18.9	
h_S	(K Ft)	152	
No. Engines		5 × 1,207	3 SSME
Eng Thrust, SL	(K Lb)	1,196	365
VAC	(K Lb)	1,376	472

22SV17986B

The major system parameters describing the selected 120-inch solid rocket motor (SRM) parallel burn system are shown. The external oxygen-hydrogen tank (EOHT) in this system is of monocoque construction. The system incorporates growth margins of 10 percent in the orbiter, 2 percent in the EOHT, and 0 percent in the 120 SRM's because they are currently developed motors. An additional 10-percent growth margin in orbiter and attached structure is available.

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System Size Selection

156 in. Parallel Burn

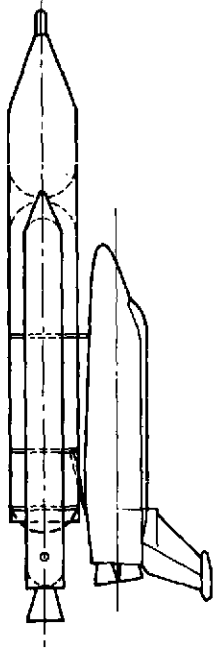| Cargo Bay Payload (K Lb)		15 Ft × 60 Ft 40K Polar	
		Booster	Orbiter
System GLOW	(K Lb)	4,898	
T/W at Lift-off		1,423	1.44
Stage GW	(K Lb)	3,112	1,785
Ascent Prop.	(K Lb)	2,740	1,392
Stage Dry Wt	(K Lb)	342	149.9
Stage Landing Wt	(K Lb)		194.3
External Tank			Monocoque
External Tank Wt	(K Lb)		73.5
q Max	(PSF)	653	
V Staging	(FPS)	5,333	
q Staging	(PSF)	43	
γ Staging	(Deg)	26	
h Staging	(K Ft)	155	
Engines	(K Lb)	2×2.94(SL)	3×472(VAC)

225V18080C

The major system parameters for the selected 156-inch solid rocket
motor (SRM) parallel burn system are shown. The external oxygen-
hydrogen tank (EOHT) features monocoque construction. The growth mar-
gins built into the initial sizing include 10 percent in the orbiter, booster,
and attached structure and 2 percent in the EOHT. An additional
growth margin capability equivalent to 10 percent in the orbiter and
attached structure has also been included.

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Booster

ALTERNATE BOOSTER
(2) 180 IN. MOTORS
3,190 M LB THRUST EACH

GLOW	4.907 M LB
BLOW	3.630
OLOW	1.277 M LB
V_{STAGE}	5,300 FPS

(2) 156 IN. MOTORS
3,190 M LB THRUST EACH
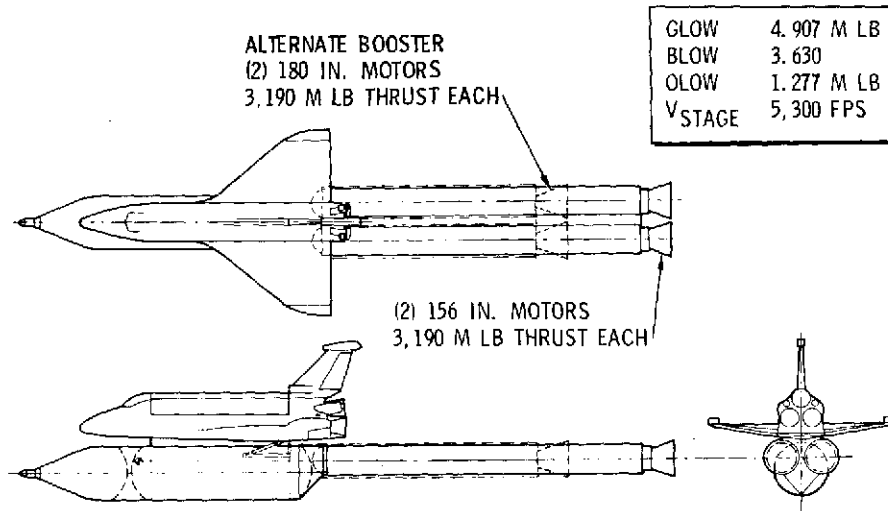
22SV18139B

A series burn tandem arrangement configuration employing the baseline
orbiter with two 156-inch solid rocket motors (SRM's) was developed.
The system had a gross liftoff weight (GLOW) of 4.907 million pounds
at a staging velocity of 5300 fps. A review of the configuration revealed
that the length to diameter (l/d) ratio of the SRM's was excessive. Therefore,
the configuration was modified to reflect two 180-inch-diameter motors
that provided an acceptable l/d and resulted in the configuration shown.

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| Orbiter 15 Ft × 60 Ft (65U/40D) | Orbiter 14 Ft × 45 Ft (45U/25D) | SERIES BURN RECOVERABLE PFB | | Orbiter 15 Ft × 60 Ft (65U/40D) | PARALLEL BURN PFB |

	15 × 60	14 × 45
GLOW (M Lb)	5.84	6.66
Prog Cost (B)	9.41	Δ -0.36
DDT&E Cost (B)	4.59	Δ -0.17
Cost/Flt (M̄)	7.6	Δ -0.35

GLOW (M Lb)	6.56
Δ Prog Cost (B)	+0.79
Δ DDT&E Cost (B)	+0.03
Δ Cost/Flt (M̄)	+0.3

| Orbiter 15 Ft × 60 Ft (65U/40D) | SERIES BURN RECOVERABLE F-1 |

GLOW (M Lb)	5.28
Δ Prog Cost (B)	-0.49
Δ DDT&E Cost (B)	-0.39
Δ Cost/Flt (M̄)	-0.1

Note: Delta Costs Referenced to Series Burn Recoverable PFB, 15 × 60 Cargo Bay

22SV1820?B

The gross liftoff weight (GLOW) and programmatic cost data are summarized as shown for the liquid-fed booster systems investigated in this study. The baseline system is the series burn recoverable pressure-fed system with the 15- by 60-foot cargo bay (65 up/40 down payload requirement). All other costs are referenced to this baseline. The series burn recoverable F-1 system shown has briefly been discussed previously in this report. Sizing of this system resulted in a gross liftoff weight (GLOW) of 5.28 million pounds with some savings in program cost; design, development, test, and evaluation (DDT&E) cost, and cost per flight compared with the baseline. For the liquid-fed recoverable systems, parallel burns result in somewhat higher GLOW's and higher costs. These increases are attributed to the poor mass fraction that results as the pressure fed systems decrease in size.
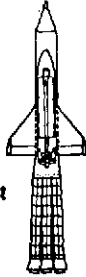
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Sys Size & Cost Summary-SRM Boosters

SERIES BURN 1207 SRM'S

1207 PARALLEL BURN (MONOCOQUE)

Orbiter
15 Ft × 60 Ft
(65U/40D)

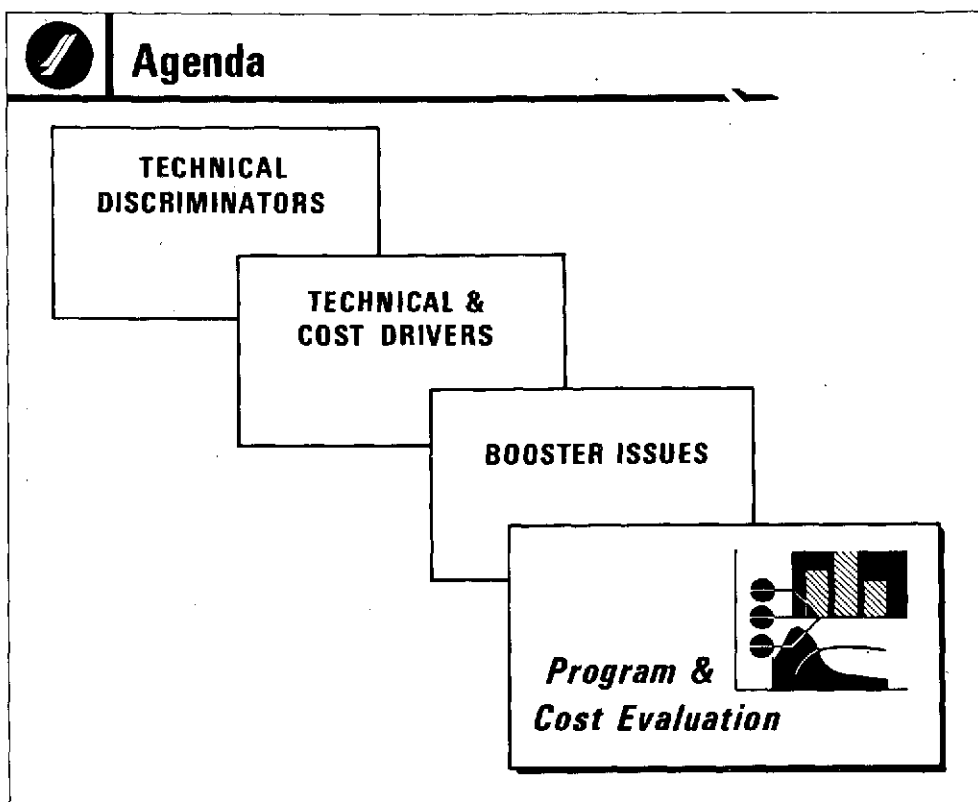Orbiter
15 Ft × 60 Ft
(65U/40D)

Orbiter
14 Ft × 45 Ft
(45U/25D)

	6 SRM Skin-Str	7 SRM Mono
GLOW (M Lb)	5.43	6.11
Δ Prog Cost (B)	+3.79	+4.87
Δ DDT&E Cost (B)	-0.74	-0.86
Δ Cost/Flt($\overline{M}$)	+11.5	+13.8

	15 × 60		14 × 25
	5	6	4
GLOW (M Lb)	524	5.83	4.39
Δ Prog Cost (B)	+2.71	+3.78	+1.42
Δ DDT&E Cost (B)	-0.95	-0.89	-1.11
Δ Cost/Flt ($\overline{M}$)	+9.3	+11.6	+6.6

Note: Delta Costs Referenced to Series Burn Recoverable PFB, 15 × 60 Cargo Bay

22SV18201B

As on the previous chart, all costs shown are referenced to a series burn recoverable pressure-fed booster (PFB) 15- by 60-foot cargo bay system. Shown are the gross liftoff weights (GLOW's) and programmatic costs for the series burn 1207 solid rocket motors (SRM's) and the parallel burn 1207 SRM's. As shown previously, the program costs for the SRM's and the cost per flight are substantially higher than for the liquid fed systems. The design, development, test, and evaluation (DDT&E) costs, however, are significantly lower. These conclusions also apply to the 156-inch SRM systems.

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Agenda

TECHNICAL DISCRIMINATORS

TECHNICAL & COST DRIVERS

BOOSTER ISSUES

Program & Cost Evaluation

22SV18071C

In this section, costs are compared to provide an evaluation of the various systems previously discussed and the program funding required.

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Cost Comparison - Series Boosters

15 Ft x 60 Ft Cargo Bay Size
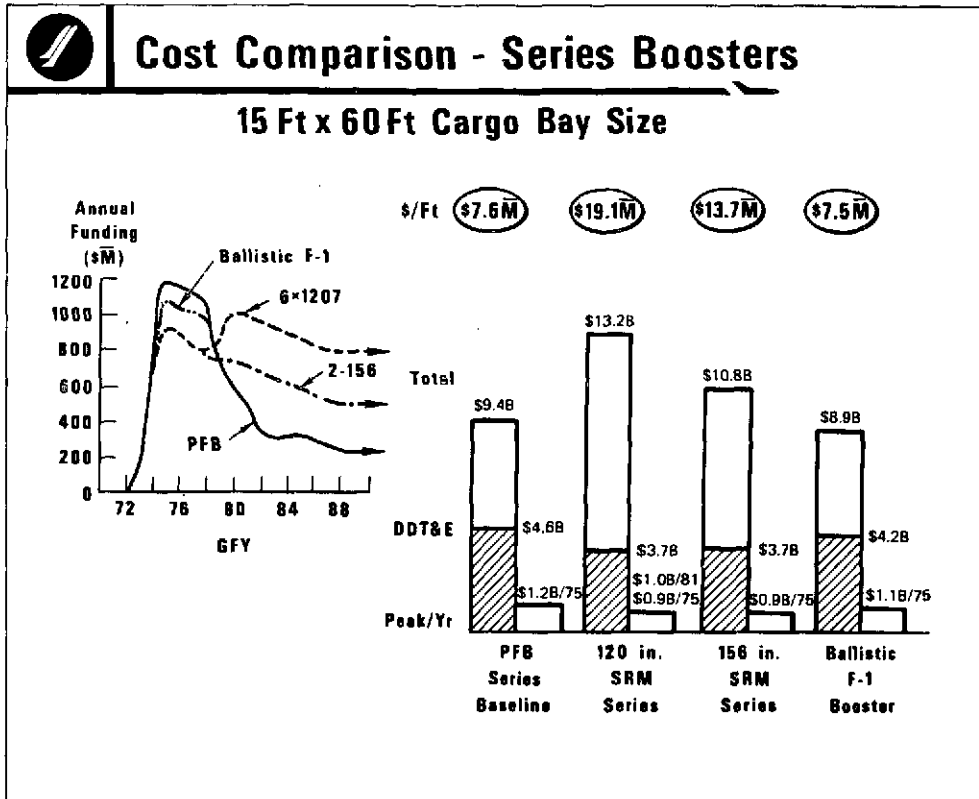
Program funding requirements and cost comparisons for the series systems
are shown. The liquid pressure-fed booster (PFB) 120-inch solid rocket
motor (SRM), 156-inch SRM, and ballistic F-1 booster systems are com-
pared. The liquid-fed systems have the lowest cost per flight and the lowest
program costs. The 120-inch SRM's have the highest cost per flight and
no particular advantage over the 156-inch SRM systems in other cost
categories. Therefore, the 120-inch SRM series system should be dropped
from further consideration. The ballistic F-1 booster is attractive com-
pared with the PFB series system, but little design analysis is available
at this time to substantiate the cost figures. The system, however, is
attractive enough to warrant further investigation. The two attractive
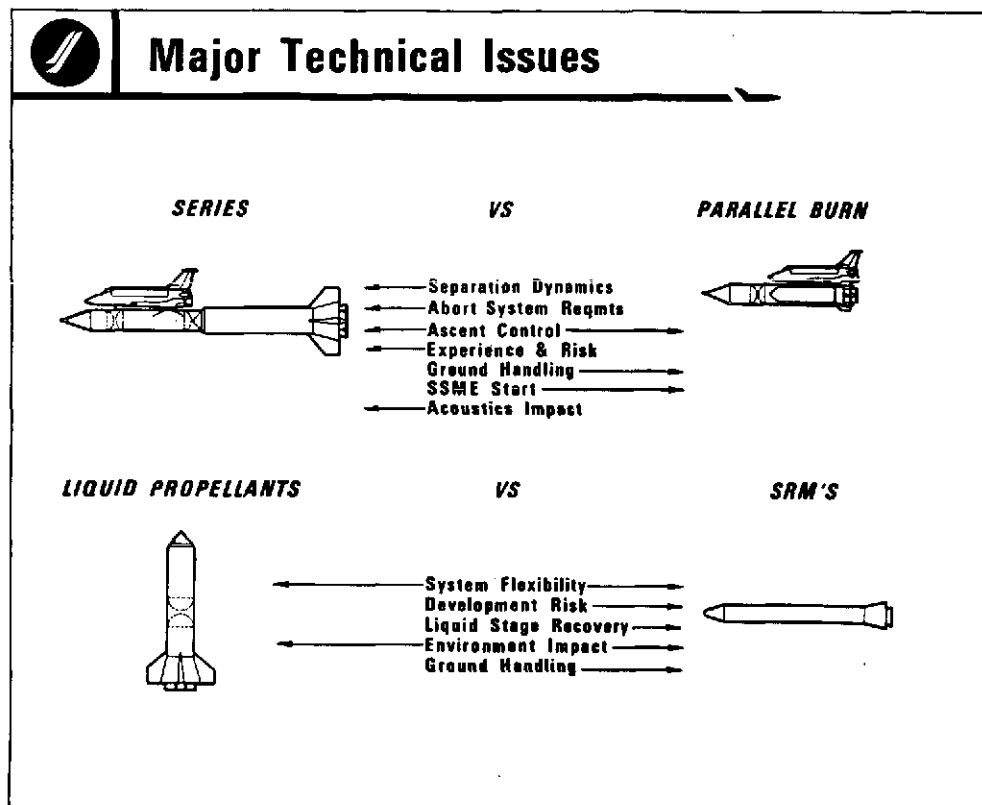systems then are the PFB series baseline and the 156-inch SRM system.

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Cost Comparison - Parallel Boosters

15 Ft x 60 Ft Cargo Bay Size
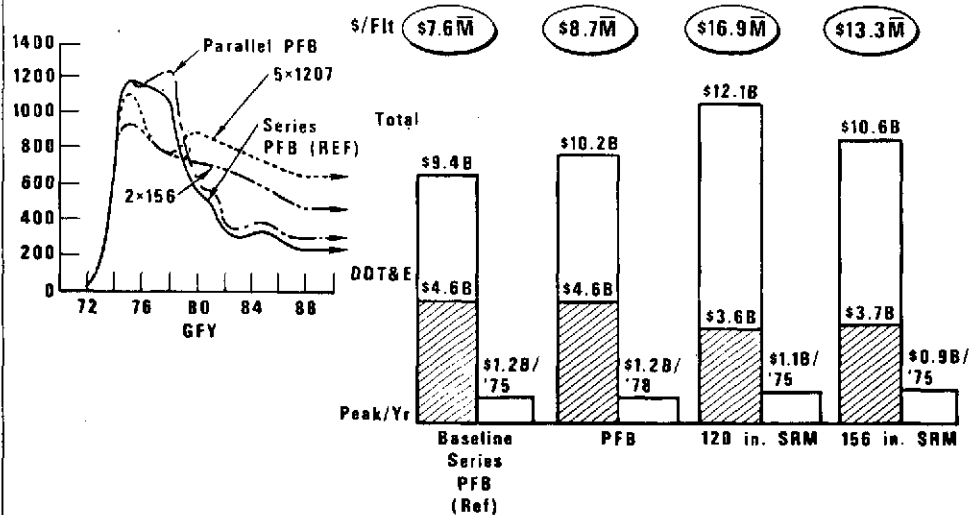
Programmatic costs have been computed and are compared for the parallel systems considered in this study. Again, on the cost per flight basis, solid rocket motor (SRM) systems are much more expensive than the liquid-fed systems. Of the two parallel SRM booster systems, again, the 156-inch SRM system is the more attractive and likewise is more attractive than the parallel burn pressure-fed booster (PFB) because of its lower design, development, test, and evaluation costs. Therefore, of the three parallel systems considered, the 156-inch SRM system should be retained for further comparison. The PFB parallel burn system has no advantage over the series burn, and it should therefore be deleted from further consideration.

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Major Technical Issues

SERIES   VS   PARALLEL BURN

Separation Dynamics
Abort System Reqmts
Ascent Control
Experience & Risk
Ground Handling
SSME Start
Acoustics Impact

LIQUID PROPELLANTS   VS   SRM'S

System Flexibility
Development Risk
Liquid Stage Recovery
Environment Impact
Ground Handling

22SV18263C

The major technical issues described in this report are summarized on the associated chart. It has been shown that the separation dynamics related to the series system are significantly more straight forward than for the parallel burn system. Likewise, implementing the abort system in terms of solid rocket motor (SRM) boost again is simplier because of the lack of a thrust vector control (TVC) requirement on the ASRM motors. Ascent control requirements for both systems are approximately the same; however, the key issue of whether or not TVC is required on the booster motors in the parallel burn system has been resolved as follows: Unless significant impact to the orbiter is accepted, booster TVC is required for the parallel burn systems. Background experience and relative risk favor the series systems because of a long history of successful series burn launch vehicles. The parallel burn system appears to have some ground handling advantages because of its close proximity to the ground. Some advantage is seen for the parallel burn system in that the space shuttle main engine (SSME) motors are started on the ground, which gives assurance that these engines are running stably before liftoff. With regard to acoustics, no significant difference as far as impact on the ground is concerned is seen between the two, but a somewhat higher impact on the parallel burn system compared with the series burn system is seen. It is felt that the liquid propellant system is somewhat more flexible than the SRM's because its ability to tailor the thrust time history at almost any point in the program. The development risk, however, appears to favor the SRM's because of their greater simplicity. It is felt that recovery of the liquid-fed, pressure-fed booster constitutes a significant program risk. Neither system has significant impact on the ground environment. The ground handling for the SRM's appears to be somewhat simplier than for the liquid propellant systems. Based on technical merit, it is recommended that the accepted system incorporate a series burn mode using solid propellant motors.

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Cost Comparison Major Candidates
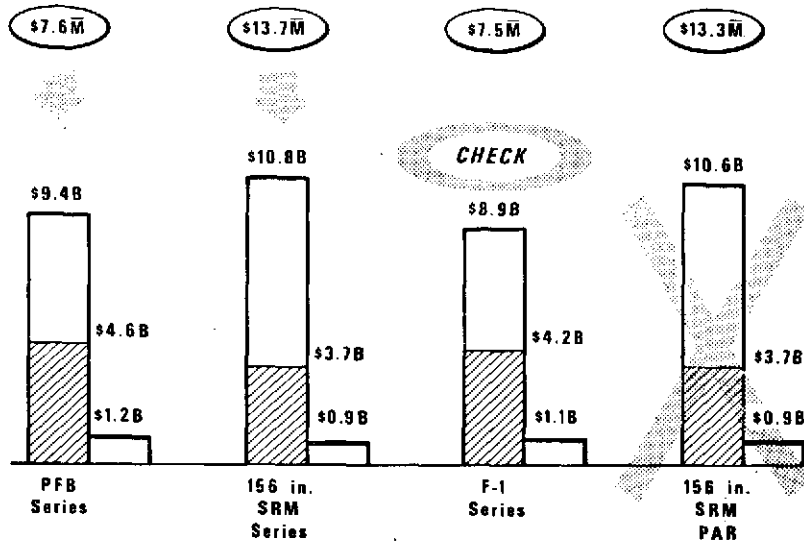
Programmatic cost comparisons are shown for the most attractive of the systems considered. In a comparison of the 156-inch solid rocket motor (SRM) systems, the parallel burn system has no significant advantage over the series burn system, but as shown previously, the series burn system is preferred technically. Therefore, it is recommended that the 156-inch SRM parallel burn system be dropped from consideration. The F-1 series burn recoverable boost system is attractive; however, more investigation is needed to verify the technical merits of this sytem and the cost predictions. Finally, the pressure-fed booster (PFB) series burn system compared with the 156-inch SRM series burn system has significantly lower cost per flight and programmatic cost, but its design, development, test, and evaluation peak annual funding are more than those of the 156-inch SRM series burn system.

Conclusions

- Series Comparable Cost to Parallel & Less Risk

- SRM'S Lower Devel Risk

- Liquid Boosters Best Meet all Cost Goals

- 15 x 60 Orbiter Best

- If Minimum Development Risk and/or Cost is the Major Criteria-Choose Series-Solid

- If Meeting All Cost Goals is the Major Criteria -Choose Series-Liquid

22SV182568

Conclusions of this study are shown on the associated chart. It has been shown that the series burn system have comparable cost to the parallel burn systems and have less risk from a technical viewpoint. Also, it is felt that the solid rocket motor development program entails less risk than that of the pressure-fed boost system. The survey of the cost of both the solid and liquid-fed systems compared with the program cost goals illustrates that the liquid systems best meet all the goals, although they do exceed the design, development, test, and evaluation (DDT&E) and peak annual funding limitations slightly. Major cost savings could be accrued with a reduction in the up payload, but little advantage was gained by reducing the payload bay size or the down payload. Therefore, it is recommended that the 15- by 60-foot cargo bay orbiter be retained as the baseline. Finally, if minimum development risk and minimum development cost are the major criteria for program selection, then a series burn configuration using solid propellant boosters should be selected. However, if closely approximating all cost goals is a major criterion, then a series burn system with liquid propellant should be the selected option.

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